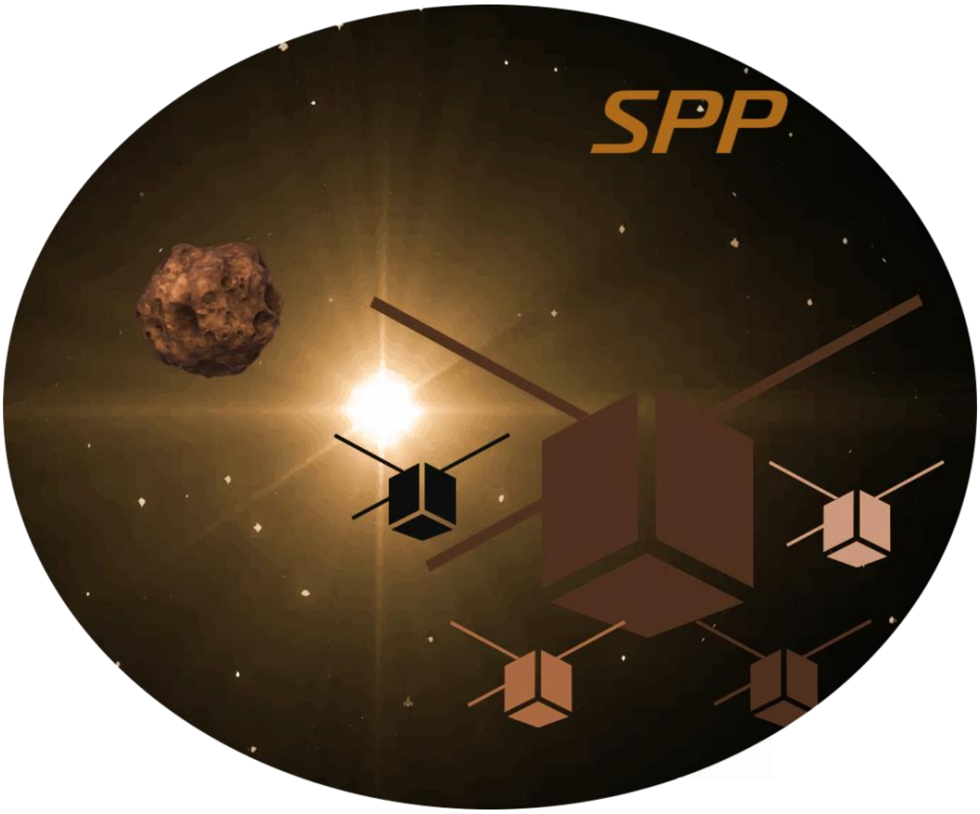
CDF STUDY REPORT

SPP EXECUTIVE SUMMARY

Consolidated Data for Small Planetary Platforms in NEO and MAB



concurrent design facility

CDF-178(C)
January 2018

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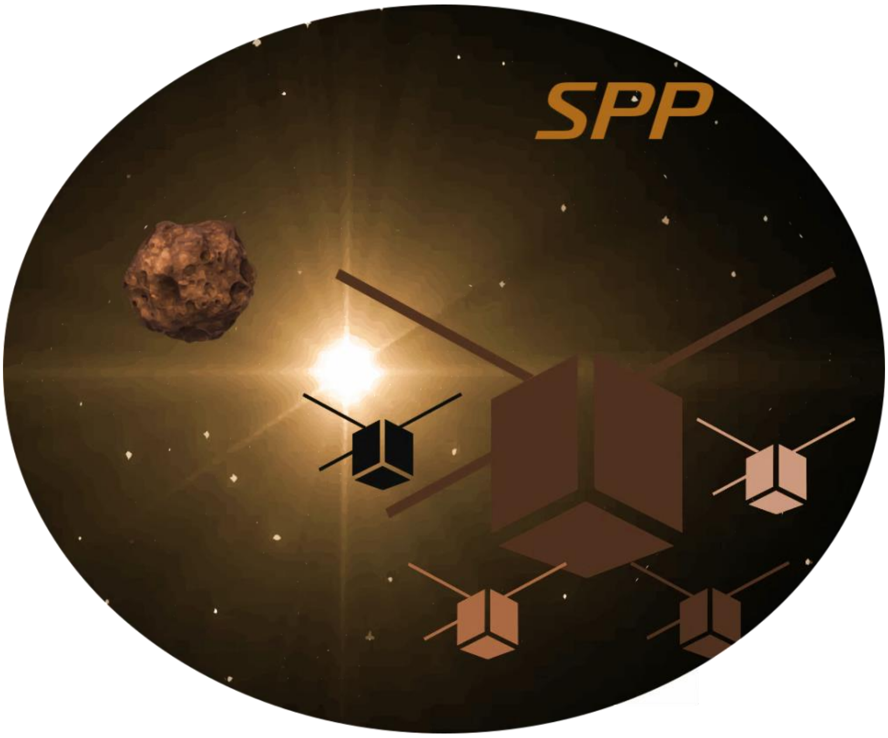

FRONT COVER

Study Logo showing satellite approaching an asteroid with a swarm of nanosats

STUDY TEAM

This study was performed in the ESTEC Concurrent Design Facility (CDF) by the following interdisciplinary team:

TEAM LEADER			
AOCS		PAYLOAD	
COMMUNICATIONS		POWER	
CONFIGURATION		PROGRAMMATICS/ AIV	
COST		ELECTRICAL PROPULSION	
DATA HANDLING		CHEMICAL PROPULSION	
GS&OPS		SYSTEMS	
MISSION ANALYSIS		THERMAL	
MECHANISMS			

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With the technical support of:

Systems/APIES

Smallsats

Radiation

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1 INTRODUCTION

1.1 Background

Requested by SCI-FM and financed by GSP, the CDF Small Planetary Platforms (SPP) study carried out an assessment of small planetary mission concepts including a mothercraft and a swarm of smallsatellites. The study was organised in 8 design sessions, starting with a Kick Off on the 8th November 2017 and ending with an Internal Final Presentation on the 6th December 2017. An additional session with a reduced number of specialists took place at the end of January 2018 to look into the concept of a multi-asteroid tour with small satellites. The design team consisted of a multidisciplinary team of experts and included input from science and other directorates.

The concept studied was a proposal to perform multi-point (and possibly multi-target) measurements around small bodies (asteroids and comets), as well as Mars or Venus allowing the scientific community to gather information from different locations simultaneously. The potential interest in "multi-point measurement science", has been highlighted following missions like Rosetta.

1.2 Objective

The main goal was not to design a specific mission but to provide a “tool-box” of technical building blocks that the community can use to develop new planetary missions architectures, in reply to future science calls.

The objectives of the SPP study was to:

- Assess the feasibility of performing deep space planetary missions with an architecture consisting of a mothership spacecraft carrying a swarm of smallsats to be deployed for multi-point science observations.

1.3 Scope

The scope of the study was very wide ranging and rather than follow the traditional CDF study concept of trying to reduce the options and then studying a small number of them in detail, this study expanded the options to try to increase the potential usage of the toolbox.

- Highlight the main operational constraints (i.e. max communication range vs achievable data rates, communication links between the mothership and the swarm, max number smallsats, etc.) imposed by the architecture, identifying technical solutions for a variety of scenarios including rendez-vous missions to small bodies, as well as missions around Mars and Venus.
- Identify any new specific technology developments enabling missions.
- Preliminarily design the mothercraft and the smallsats and perform parametric analysis to understand the flexibility/adaptability of the design to various environments.
- Assess the possibility of adding a lander asset on the surface of the small body.

- Provide a portfolio of potential transfers to small bodies for launches between 2024 and 2034.
- Define the programmatic approach, including the procurement of the smallsats as part of the payload complement.
- Assess the mission cost, with a target of 150M€ (i.e. fit in an "F class").

1.4 Document Structure

The layout of this report is different to a standard CDF Study, in that there are 3 main reports, one covering SPP for NEO Inactive Bodies, one covering SPP for Main Asteroid Belt Active Bodies (CDF-178(B)) and an Executive Summary (this document), that compiles the main aspects of the two documents, the system-level and main sub-system level trade-offs and covers the top level synthesis (CDF-178(C)). Details of the study results can be seen in the Table of Contents. The details of each domain addressed in the study are contained in specific chapters.

Due to the different distribution requirements, only cost assumptions excluding figures are given in this report. The costing information is published in a separate document

Note: In the drawings and figures included in this report sometimes the acronym NS is used to refer to the smallsats. NS and SS should be understood as one and the same thing.

2 MISSION OBJECTIVES

2.1 Background

A call for New Science Ideas was issued in 2016 to invite the scientific community to propose ideas and topics for future science missions without addressing a specific mission. Three "themes" were selected as an outcome of that call, one of them being "Planetary science missions vs. platform size" aiming at exploring options for implementing planetary missions with small-class satellites.

At a workshop organised in September 2017, members of the European and Japanese planetary scientific community discussed possible scenarios and the best approach for a study. The main outcome of the workshop was the interest of the scientific community in studying a "multi-point" (simultaneous) observations mission in which a "swarm" of small satellites is placed around a small body and can fly close to its surface. This outcome stems as one of the Rosetta science lessons learnt. Further options for lander(s), multi-target observations or even investigations around Mars (Phobos/Deimos) and Venus were also deemed interesting.

After some iterations with the community on the initial concept, the following two reference scenarios were selected for study:

- A radar tomography mission around an inactive Near Earth Asteroid (NEA)
- A volatiles investigation mission around an active asteroid in the main asteroid belt.

Given that the CDF study M-ARGO had already assessed the feasibility of flying to a NEO with a small-class satellite (a cubesat in fact), it was decided that the SPP study would consider an architecture in which a mothercraft, not carrying any scientific instrument itself, carries the flotilla of small satellites to the vicinity of the selected body and also performs the data relay function back to Earth.

The Small-satellite design can be fully customised for the specific payloads and environment. In the frame of this study, it was decided to adopt the standard cubesat form factor (with its limitations) in order to make use, as much as possible, of existing cubesat technology e.g. the deployer mechanism. This approach can however be reassessed once a detailed mission design is proposed.

2.2 Mission Justification

The two selected mission reference scenarios should be representative enough to size the mother/daughters architecture and to understand its capabilities regardless of the final selected target (within a determined set of boundary constraints). A high level assessment to judge the adaptability of the concept/architecture to missions involving landers, missions around planetary bodies (mainly Phobos) and multi-target missions was also required.

2.3 Science Objectives

When the Solar System was formed, planetesimals constituted the building blocks of protoplanets and eventually of the planets themselves. Asteroids and comets are the remainders of that early stage. Therefore, exploring their population is a key to understanding the Solar System's history and evolution. There are multiple ways of studying these small bodies and for the purpose of this study two science focused themes were selected, always from the "multi-point" measurement perspective which would be the new feature with respect to past missions.

The science objective of the mission to an inactive body in the NEO range would be to study the body's interior structure by means of tomographic measurements. Knowledge of the interior structure is of great importance for Earth impact models, and crucial to find reliable ways to deflect asteroids which are a threat to Earth. In addition, remote sensing of the surface topography and distribution of morphological features (e.g., boulders, craters, fractures) provides valuable historical information. The focus of the science observations is on probing the inside and the near surface with radars, with supporting visual and IR imaging, to characterise the surface composition. The use of low frequency radars to carry out bi-static type measurements needs more than one spacecraft (ala CONSERT on Rosetta/Philae) and can really benefit from the simultaneous measurements from at least two spacecraft. More that one camera allow for stereophotogrammetry, better to resolve 3D and also the phase dependence and characterisation of the surface of materials. Multi-point observations in this case should also help in retrieving a shape model faster, more efficiently, and any changes observed.

For the reference scenario of a mission to an active body in the main asteroid belt the science theme that was selected for the purpose of the study is the spatial and temporal evolution of dust and volatile material. The instruments on Rosetta recorded considerable variation in the abundance and nature of dust, volatiles and organics in the coma around comet 67P. It is known that specific areas on the body's surface emit larger amounts of material than others. However, understanding the variations between different regions, and how these are evolving with diurnal heating, orbital position and variations in surface composition/topography, etc. remain poorly understood. In this case multiple points of simultaneous observations give a large improvement on spatio-temporal changes. Rosetta was limited in that it could only sample in situ one point and having more satellites would have improved the science greatly. The same argument applies for plasma instruments for which more measurement points get rid of spatio-temporal ambiguities. For the cameras, images of the outgassing and dust at different phase angles simultaneously allow for ~3d structure.

3 MISSION OPTIONS

Two main mission options were considered with the aim of developing a general ToolBox, that would help in identifying design drivers and major considerations for missions to targets ranging from Near Earth Objects to Asteroids or Comets in the Main Asteroid Belt (MBA). The system shall be composed of a mother spacecraft and at least 4 smallsats. The Mother spacecraft should be able to perform the transfer to the target in less than 5 years from the launch date and act as a communication relay at the target. Each of the smallsats shall be able to accommodate at least 3 kg of payload.

To build this toolbox, first a system trade-off was made between target, launcher and propulsion strategies. The main factors to be compared are payload mass for each smallsat at target and the transfer time.

It has to be emphasised that the trajectories analysed during the study have been optimised assuming the constraint of launching together with ARIEL to L2 and then departing from there to reach the NEO target (Option 1) or the Main Asteroid Belt target (Option 2). Many other transfer possibilities exist depending on a different launch strategy.

3.1 Target Selection

Figure 3-1 helps to visualise the vast trade space that was studied during this CDF, in terms of range of heliocentric semi-major axis of the targets envisaged. Apart from Phobos as a bonus option, all of the considered targets are called small bodies. For this CDF study, objects are called asteroid or comet, depending on which database they are listed by the International Astronomical Union (IAU) on the Minor Planet Center website RD[1].

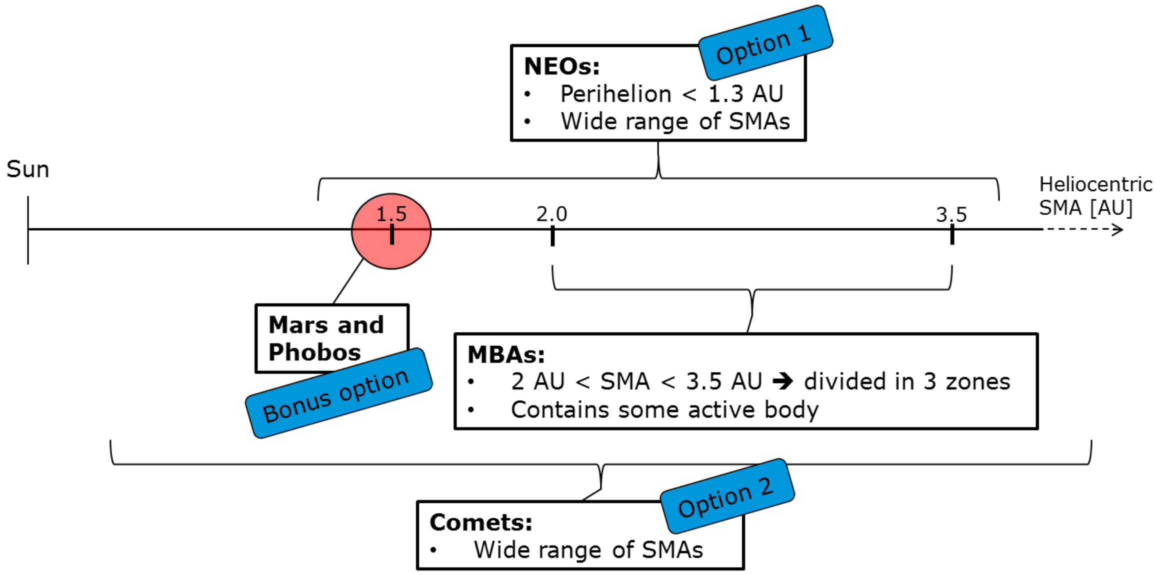


Figure 3-1: Different scenarios considered during the SPP CDF study

Some examples of possible targets considered in this study are listed in Figure 3-2.

The transfer orbit and related transfer times to these different targets were assessed. Finally it was decided to focus the study on NEOs and the Asteroid Belt and the Martian moons in order to have the most benefits of a distributed measurement of a swarm of satellites.

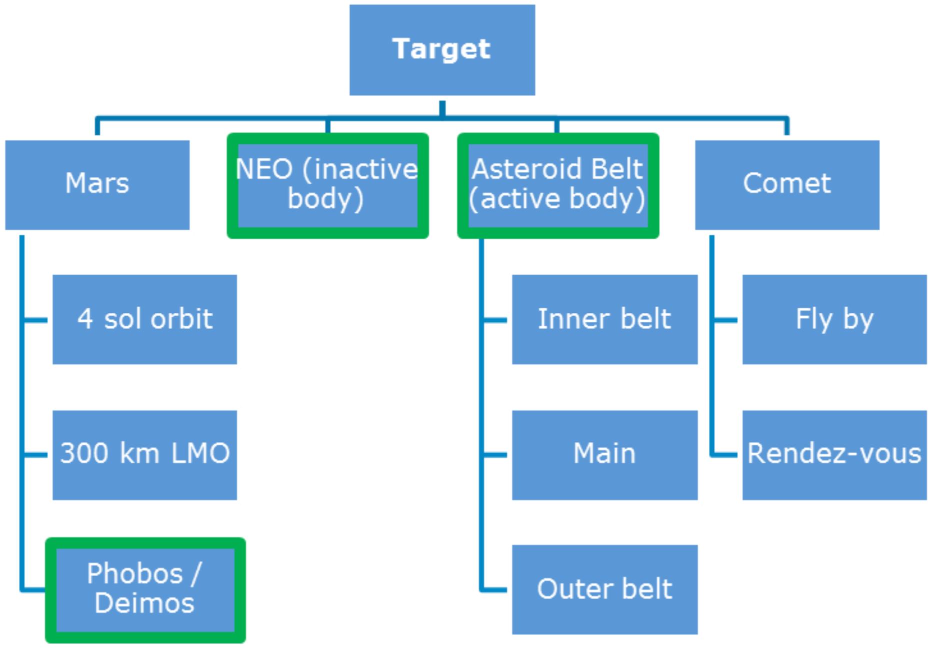


Figure 3-2: Overview of possible Targets. Green framed targets are most interesting for distributed measurements

Regarding the targets in the Asteroid Belt, following the initial assessment based on the delta-V and propulsion architectures (toolbox), it was decided to focus the study on asteroids from the inner belt in order to limit the overall system mass and complexity.

The trajectory analysis and optimisation was done considering launch windows in the 2024-2034 range. This period was assumed in order to maintain compatibility with the next M-class missions. ARIEL appears particularly interesting since it will fly to L2 and has a launch margin of about 1/3 of the total launcher (AR 6.2) capacity to L2, making it ideal for considering a co-passenger.

The arrival date with respect to the target perihelion was also a parameter considered in the trajectory determination and would have to be further assessed depending on the science impact of the target position with respect to the Sun at the time of performing the scientific measurements.

3.2 Launcher Options

To perform the insertion of the whole system, several launchers have been considered. The launchers that have been analysed are the Japanese Epsilon, the VEGA-C and a shared Ariane 6.2. The Epsilon and Vega-C are considered to launch into a Low earth orbit, while the Ariane 6.2 could launch either into a GEO transfer orbit or to the second Lagrange point (L2). Soyuz was discarded because it would not be available for a launch from Kourou in the given timeframe. An overview of the considered launchers can be seen in Figure 3-3.

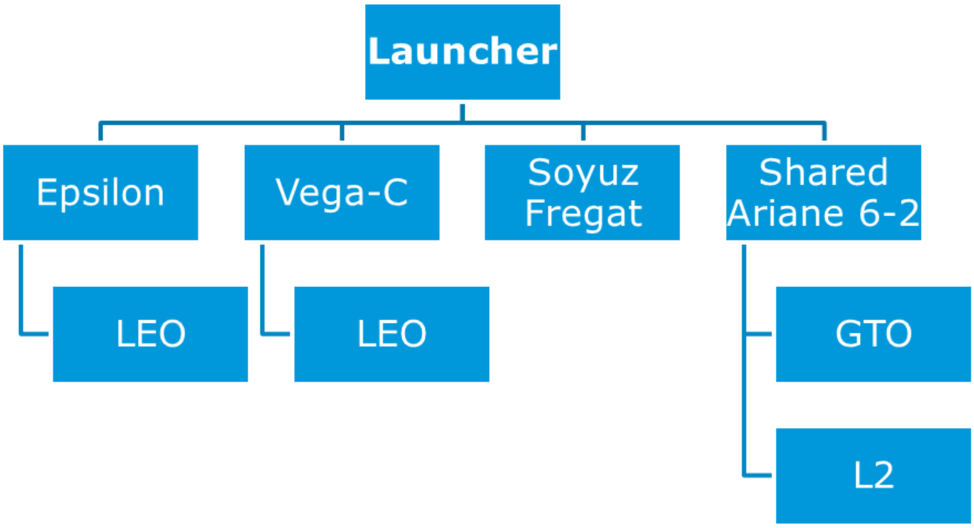


Figure 3-3: Launcher Overview

3.3 Propulsion Architecture

Several propulsion architectures for the mother spacecraft have been traded-off. The technologies are based on either chemical or electrical propulsion. Each of the two technologies have their advantages and disadvantages.

The electrical propulsion systems have in general a higher specific impulse (ISP) which means a higher efficiency in terms of propellant mass needed. On the other hand, the thrust is very low, which results in a higher time of flight.

The chemical propulsion system has a higher thrust, which means very fast changes in velocity and therefore close to optimised manoeuvres, resulting in a shorter time of flight. However the lower ISP leads to significantly higher need of propellant mass.

The options for this trade-off are 1) a combination of a chemical kick-stage and a chemical mother spacecraft, 2) a combination of a chemical kick-stage and an electrical mother spacecraft or 3) an electric propulsion mother spacecraft. An overview of the different propulsion architectures considered in Figure 3-4. Further details on the propulsion trade-off for the mother spacecraft are provided in paragraph 3.6.1.

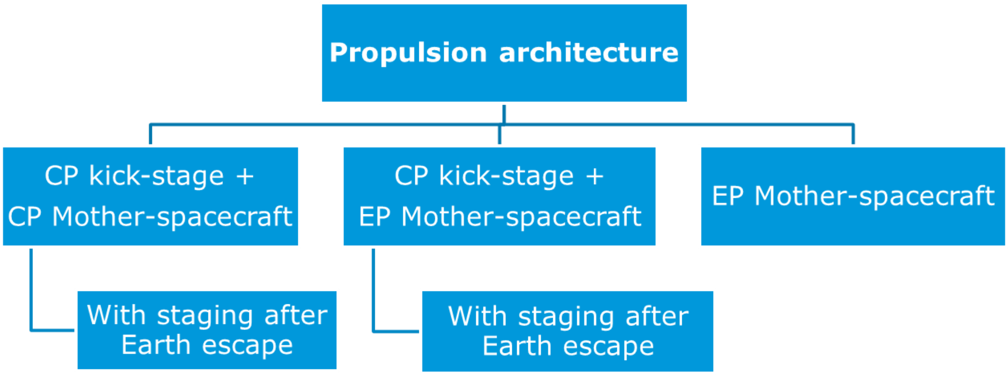


Figure 3-4: Overview of propulsion architecture

3.4 Architecture Trade-Off Assumptions

Several assumptions were made to trade-off the different combinations of Target, Launcher and Propulsion architecture. Table 3-1 indicates the assumption made for the mass to target orbit for each launcher. Since the Ariane 6.2 can launch into two orbits as a shared launch, the available payload mass figure was considered excluding the necessary Sylda-like adapter (including margin).

Launcher	Insertion Orbit	Payload mass (kg)	Comment
Epsilon	LEO	1200	Payload to 250x500 km orbit
Vega-C	LEO	2200	Payload to 700x700 km orbit (not optimised)
Shared Ariane 6.2	GTO	2000	A62 target performance to GTO 5000kg 2000kg assumed for shared launch excluding 800kg Sylda-like adapter + 100kg margin
Shared Ariane 6.2	L2	900	A62 target performance to L2 2800kg 900kg assumed for shared launch excluding 800kg Sylda-like adapter + 100kg margin

Table 3-1: Launcher performance assumptions

For the performance of any chemical system used in the trade-off, the following assumptions, shown in Table 3-2, have been made based on the Lisa Pathfinder propulsion module.

Type	Parameter	Value	Comment
Chemical propulsion	Thrust	400 N	Apogee engine
Chemical propulsion	ISP	325 s	Bi-propellant

Table 3-2: Chemical propulsion performance assumptions

As a reference for the electrical propulsion system, the two types of engines have been considered, the Kaufman-type Gridded Ion Thruster (for which T6 was used as a reference) and the Hall Effect Thrusters (for which the reference was PPT1350). These can be seen in Table 3-3.

Type	Parameter	Value	Comment
T6	Thrust	0.145 N	Maximum thrust considered, modulated with thrusting time
T6	Average ISP NEO	4000 s	Considered that maximum power is available for whole transfer
T6	Average ISP Asteroid Belt	3500 s	ISP reduced with available power

Type	Parameter	Value	Comment
PPT1350	Thrust	0.08 N	Maximum thrust considered
PPT1350	ISP	1640 s	Considered that maximum power is available for whole transfer

Table 3-3: Electric propulsion performance assumptions

In order to compute the transfer time with the electrical propulsion systems, several assumptions on the thrusting time have been made. These highlight the fact that the propulsion system is not firing during the entire transfer, only for a fraction of it. These fractions are very dependant on the trajectory of the transfer and can vary a lot. For a more detailed and mature estimation a numerical evaluation would be necessary. The assumptions presented in Table 3-4 are meant as a high level estimate to allow carrying out the system level trade-off.

Type	Parameter	Value	Comment
T6	Thrusting time – orbit raise	80%	10% contingencies + 10% eclipses
T6	Thrusting time – transfer NEO	60%	
T6	Thrusting time – transfer Asteroid Belt	40%	Accounts also for thrust level variations
PPT1350	Thrusting time – orbit raise	80%	10% contingencies + 10% eclipses
PPT1350	Thrusting time – transfer NEO	90%	50% higher to account for thrust level difference
PPT1350	Thrusting time – transfer Asteroid Belt	60%	50% higher to account for thrust level difference

Table 3-4: Electric propulsion thrusting time modulation assumptions

The assumptions for the mass ratios of payload and structure can be seen in Table 3-5. They are dependant on the available power during the transfer and the needed delta-V.

Type	Parameter	Value	Comment
Kickstage	Structural Index	15%	Dry mass of kick stage as fraction of the wet mass Reference: Lisa Pathfinder Propulsion Module 17%
Mother SC	Structural Index	15%	Fraction of dry mass
Mother SC CP	Payload mass fraction	20%	Used for configurations with CP delta-V below 7000 m/s (single power string) i.e. from L2 to NEO for all options with kick-stage Reference: Mars Express P/L mass 27%
Mother SC EP	High payload mass fraction	20%	Used for all other EP cases

Type	Parameter	Value	Comment
Mother SC EP	Low payload mass fraction	12%	Staging considered
SmallSats	High payload mass fraction	15%	Payload mass fraction per smallsat considering maximum power resources available at Asteroid Belt
SmallSats	Low payload mass fraction	12%	Payload mass fraction per smallsat considering maximum power resources available at NEO (need deployable radiator)
SmallSats	Structure mass	2.25kg	Reference for 16U https://www.isispace.nl/product/16-unit-cubesat-structure/

Table 3-5: Mass Fractions assumptions

To leave Earth with a certain V infinity, the needed delta-V is depending on the initial orbit and the propulsion system used. This is shown in Table 3-6 for the assumed initial orbits ranging from a V infinity between 1000 m/s and 6000 m/s. These values will be used as a look-up table for Table 3-7 to calculate the total Delta-V needed for the escape from Earth.

Earth	Chemical Propulsion			Electrical Propulsion		
V infinity (m/s)	LEO (m/s)	GTO (m/s)	L2 (m/s)	LEO (m/s)	GTO (m/s)	L2 (m/s)
1000	3360	1360	1000	8000	4900	1000
2000	3570	1380	2000	9000	5900	2000
3000	3850	155	3000	10000	6900	3000
4000	4230	1930	4000	11000	7900	4000
5000	4700	2500	5000	12000	8900	5000
6000	5275	3220	6000	13000	9900	6000

Table 3-6: Delta-V needed from initial Orbit to reach V infinity with CP or EP

			Chemical Propulsion				Electrical Propulsion			
Range (AU)	**Target - Orbit**	**V inf at Earth**	**LEO 700x700 (m/s)**	**LEO 250x 250 (m/s)**	**GTO (m/s)**	**L2 (m/s)**	**LEO 700x700 (m/s)**	**LEO 250x 250 (m/s)**	**GTO (m/s)**	**L2 (m/s)**
	Mars – 4 Sol	3000	3850	3680	1561	3000	10000	9200	6900	3000
	Mars – 300 km LMO	3000	3850	3680	1561	3000	10000	9200	6900	3000
	Mars Phobos	3000	3850	3680	1561	3000	10000	9200	6900	3000
Perihelion < 1.3	Neos	5000	4705	4340	2488	5000	12000	10850	8900	5000
2 < SMA <2.5	Inner Asteroid Belt	5500	4977	4518	2834	5500	12500	11294	9400	5500
2.5 < SMA 2.8	Main Asteroid Belt	6500	5589	4888	3662	6500	13500	12219	10400	6500
2.8 < SMA 3.5	Outer Asteroid Belt	7500	6293	5278	4672	7500	14500	13194	11400	7500
	Comet Flyby	4000	4232	4000	1933	4000	11000	10000	7900	4000
	Comet RV	7000	5929	5080	4144	7000	14000	12700	10900	7000

Table 3-7: Delta-V needed to reach V-infinity at Earth for each Target

After leaving Earth with the needed V-infinity the system has to perform additional delta-V manoeuvres to reach the Target. These delta-Vs can be seen in Table 3-8. The values from Table 3-7 and Table 3-8 sum up to the total delta-V needed for the mission.

Range (AU)	Target - Orbit	Delta-V with CP	Delta-V with EP
	Mars – 4 Sol	1650	3860
	Mars – 300 km LMO	2970	6200
	Mars Phobos	2550	5000
Perihelion < 1.3	Neos	2000	2000
2 < SMA <2.5	Inner Asteroid Belt	4500	4500
2.5 < SMA 2.8	Main Asteroid Belt	5500	5500
2.8 < SMA 3.5	Outer Asteroid Belt	5500	5500
	Comet Flyby	0	0
	Comet RV	5000	5000

Table 3-8: Delta-V needed for heliocentric transfer to Target

3.5 Architecture Trade-Off Results

The overall system trade-off was made with 9 targets, 4 different launch strategies and 3 propulsion architectures. This leads to a combination of 108 possible mission options.

To evaluate these architectures, an Excel Table has been created in a pivot table (toolbox) with the above-mentioned assumptions. All masses to different mission phases have been calculated and transfer times are given for each.

For reasons of overview, only the most promising targets for distributed measurements will be discussed in this section.

Table 3-9 shows the results for the NEO targets. The option of a pure CP mother spacecraft that is launched with the Ariane 6.2 to GTO could lead to the simplest mission concept and operations, but fails to meet the required 3kg of payload mass for each smallsat.

The more interesting options are the fully EP systems launched to L2 and the hybrid CP/EP designs launched to GTO. All of them are able to deliver more than 3kg per smallsat with transfer times of about 2 years to a NEO.

Target	CP Engine	EP Engine	Launcher	Mass at Target	PL/MC	#NS	Pl/NS	Payload Mass for 1 NS	Time of Flight (Years)
NEOs	CP	-	Epsilon	92.79	20%	4	12%	0.56	2.00
NEOs	CP	-	VegaC	132.62	20%	4	12%	0.80	2.00
NEOs	CP	-	Ariane 6.2 GTO	402.61	20%	4	12%	2.42	2.00
NEOs	CP	-	Ariane 6.2 L2	43.10	20%	4	12%	0.26	2.00
NEOs	-	T6	Epsilon	864.89	12%	4	12%	3.11	4.20
NEOs	-	PPS1350	Epsilon	539.89	12%	4	12%	1.94	4.76
NEOs	-	T6	VegaC	1539.84	12%	4	12%	5.54	8.16
NEOs	-	PPS1350	VegaC	921.52	12%	4	12%	3.32	9.28
NEOs	-	T6	Ariane 6.2 GTO	1514.93	12%	4	12%	5.45	6.26
NEOs	-	PPS1350	Ariane 6.2 GTO	1015.76	12%	4	12%	3.66	7.00
NEOs	-	T6	Ariane 6.2 L2	752.95	20%	4	12%	4.52	2.10
NEOs	-	PPS1350	Ariane 6.2 L2	582.48	20%	4	12%	3.49	2.14
NEOs	CP	T6	Epsilon	165.12	20%	4	12%	0.99	2.07
NEOs	CP	PPS1350	Epsilon	153.44	20%	4	12%	0.92	2.12
NEOs	CP	T6	VegaC	235.99	20%	4	12%	1.42	2.11
NEOs	CP	PPS1350	VegaC	219.30	20%	4	12%	1.32	2.18
NEOs	CP	T6	Ariane 6.2 GTO	716.44	20%	4	12%	4.30	2.32
NEOs	CP	PPS1350	Ariane 6.2 GTO	665.78	20%	4	12%	3.99	2.54
NEOs	CP	T6	Ariane 6.2 L2	76.69	20%	4	12%	0.46	2.03
NEOs	CP	PPS1350	Ariane 6.2 L2	71.27	20%	4	12%	0.43	2.06

Table 3-9: Trade-Off result for NEOs

Table 3-10 shows the results for a body in the inner asteroid belt. The option with a pure EP system launched into L2 by the Ariane 6.2 can meet the required objectives of payload mass and transfer duration.

Also the hybrid designs using a CP based kick stage and a mother spacecraft using an EP propulsion system (with a T6 or a PPS1350) launched into a GTO orbit by the Ariane 6.2 are able to deliver slightly more mass in only ~2.6 years. The impacts of including a kick-stage are further discussed in section 3.5.1.

Target	CP Engine	EP Engine	Launcher	Mass at Target	PL/MC	#NS	Pl/NS	Payload Mass for 1 NS	Time of Flight (Years)
Main Asteroid Belt Inner	CP	-	Epsilon	37.69	20%	4	15%	0.28	2.00
Main Asteroid Belt Inner	CP	-	VegaC	49.03	20%	4	15%	0.37	2.00
Main Asteroid Belt Inner	CP	-	Ariane 6.2 GTO	157.40	20%	4	15%	1.18	2.00
Main Asteroid Belt Inner	CP	-	Ariane 6.2 L2	12.04	20%	4	15%	0.09	2.00
Main Asteroid Belt Inner	-	T6	Epsilon	761.53	12%	4	15%	3.43	6.58
Main Asteroid Belt Inner	-	PPS1350	Epsilon	449.61	12%	4	15%	2.02	6.68
Main Asteroid Belt Inner	-	T6	VegaC	1348.49	12%	4	15%	6.07	12.36
Main Asteroid Belt Inner	-	PPS1350	VegaC	764.75	12%	4	15%	3.44	12.56
Main Asteroid Belt Inner	-	T6	Ariane 6.2 GTO	1340.35	12%	4	15%	6.03	10.51
Main Asteroid Belt Inner	-	PPS1350	Ariane 6.2 GTO	842.97	12%	4	15%	3.79	10.62
Main Asteroid Belt Inner	-	T6	Ariane 6.2 L2	674.84	12%	4	15%	3.04	4.28
Main Asteroid Belt Inner	-	PPS1350	Ariane 6.2 L2	483.39	12%	4	15%	2.18	4.22
Main Asteroid Belt Inner	CP	T6	Epsilon	135.81	20%	4	15%	1.02	2.14
Main Asteroid Belt Inner	CP	PPS1350	Epsilon	116.87	20%	4	15%	0.88	2.23
Main Asteroid Belt Inner	CP	T6	VegaC	176.69	20%	4	15%	1.33	2.19
Main Asteroid Belt Inner	CP	PPS1350	VegaC	152.06	20%	4	15%	1.14	2.30
Main Asteroid Belt Inner	CP	T6	Ariane 6.2 GTO	567.16	20%	4	15%	4.25	2.60
Main Asteroid Belt Inner	CP	PPS1350	Ariane 6.2 GTO	488.10	20%	4	15%	3.66	2.96
Main Asteroid Belt Inner	CP	T6	Ariane 6.2 L2	43.39	20%	4	15%	0.33	2.05
Main Asteroid Belt Inner	CP	PPS1350	Ariane 6.2 L2	37.34	20%	4	15%	0.28	2.07

Table 3-10: Trade-Off result for inner Asteroid Belt

3.5.1 Kick-Stage Trade-Off

The use of a kick-stage was assessed starting with the investigation of options based on existing designs (Lisa-pathfinder PM, AVUM+). The possibilities of a solid propulsion kick-stage (lower Isp) and water propulsion, LOX/LH2, LOX/CH4 alternatives (higher Isp but requiring significant development effort) were also considered.

In the end, the option of using a kick-stage was discarded from the baseline scenarios. This was mostly driven by a qualitative analysis of estimated development cost for a customised kick-stage and other impacts on the system design (e.g. added AOCS modes, and complexity, functions to be performed by kick stage and added equipment, structural integrity of deployed equipment). Nevertheless the exclusion of the kick stage also results in a lower operations cost and shorter mission duration that may be important in particular for targets in the Asteroid Belt requiring a longer transfer. Hence, a detailed analysis of the impacts of including a kick stage in the system architecture should be performed once the actual mission target is identified.

3.6 Main System Trade-Offs

3.6.1 Propulsion Trade-Off

For the mother spacecraft, a trade-off was made between different electric propulsion alternatives: Hall Effect thrusters, ion thrusters and arcjets. The latter was excluded given the low Isp and total impulse limitations.

A Hall Effect thruster has the advantage of having a lower power requirement. However, it also has lower Isp and total impulse capability, resulting in the need for more redundant thrusters for missions with higher delta-V requirements (e.g. to the asteroid belt). A Hall Effect thruster, the PPS1350 thruster, was selected for the NEO mission option.

For the more demanding option to a target in the main asteroid belt, the T6 ion thruster was used in the baseline, since the T5 version would not provide high enough thrust. Additionally there is no gimbal mechanism available off-the-shelf at the moment for the T5. Due to the large DV requirement of this option and the long firing times it was decided to accommodate two T6 thrusters and a gimbal mechanism. Once the BepiColombo mission is launched and its T6 thrusters are operated one should reassess if the addition of the second T6 and the gimbal are really necessary for the SPP mission as they add significant mass, complexity and cost to the mission.

For the Reaction Control System, higher thrust will be needed, e.g. for the desaturation of the reaction wheels and safe mode. Using a hydrazine based RCS would require the inclusion of a dedicated system, therefore the use of Cold-Gas system based on the EP propellant gas was baselined.

For the SS, a cold gas thruster was also advantageous when compared to the hydrazine and PPT alternatives suited for high delta-V requirements. In particular, the hydrazine option is not available in Europe.

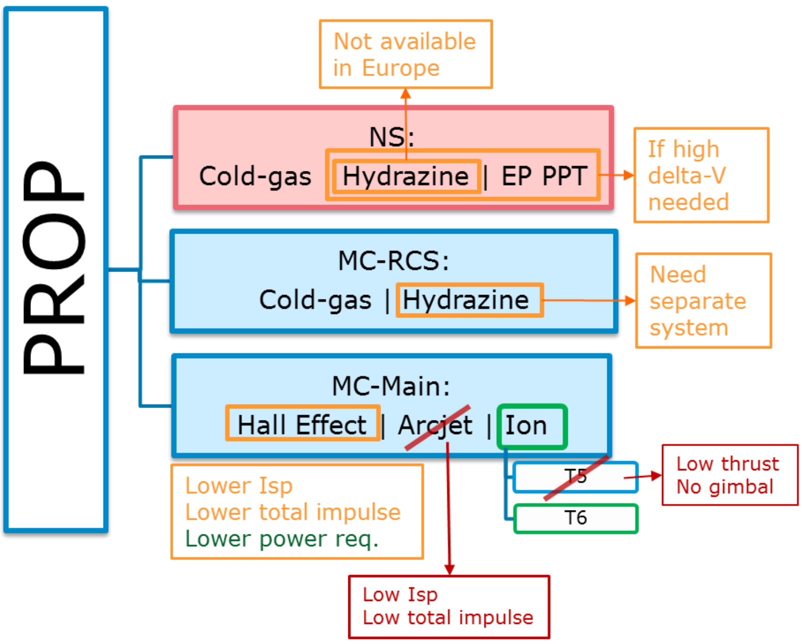


Figure 3-5: Graphical representation of propulsion trade-off

3.6.2 Communications Trade-Off

The Earth link is provided by the X-band HGA of the MC since with Ka-band the uplink of command data is not available.

For the ISL, a star architecture was selected where all the SS communicate only with the MC. Two omnidirectional S-band LGAs are required on the MC and SS allowing the MC to keep pointing the HGA to ground while communicating with the SS. The inclusion of a MGA on the MC for the ISL is considered as an option in case higher data rates are required or the distance between MC and SS increases e.g. due to constraints related to the target size (> 1 km diameter). However, it should be kept in mind that the inclusion of such antenna would require the MC to point the antenna to the SS with which it is communicating and therefore could impact the communications with ground.

A mesh architecture, where the ISL can also pass from SS to SS before transferring the data to the MC, has been highlighted as a potential enabler for certain missions profiles e.g. for large targets (> 1 km diameter). However, for the size of targets considered, this technology is not necessary as all SS can stay in visibility all the time. Moreover, being it still under development and therefore was not included in the baseline.

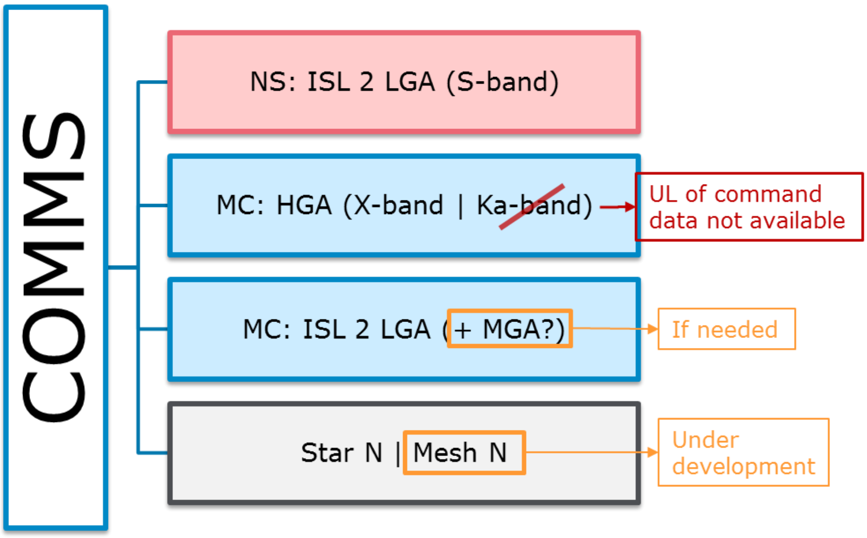


Figure 3-6: Graphical representation of communications trade-off

3.6.3 Mechanisms Trade-Off

In order to maximise power and to observe the night side of the asteroid, the SS will include a SADM.

The MC needs to include a deploying mechanism for each of the 4 SS. It was decided to adapt current designs for CubeSat dispensers, which have an important mass impact and scale up with the size and mass of the SS.

Regarding the MC, the combination of a SADM and gimbal was traded-off with a SADM and HGA pointing mechanism instead, keeping the thrusters fixed. The first option was selected as it maximises the operational flexibility in particular by reducing the RW offloading.

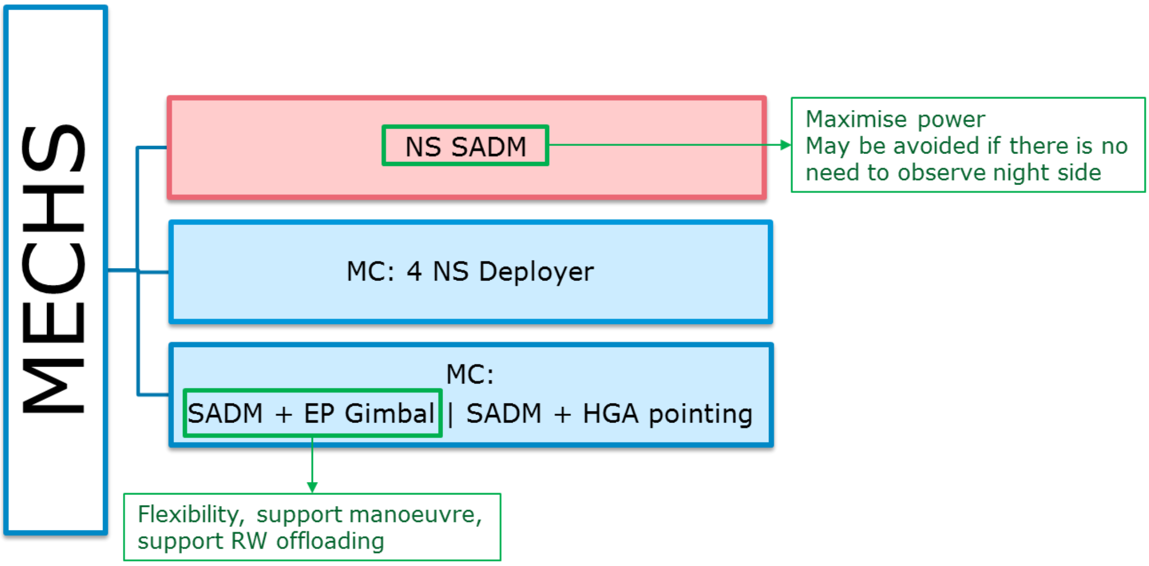


Figure 3-7: Graphical representation of mechanisms trade-off

3.6.4 Data-Handling Trade-Off

The MC DHS requires radiation-hard components following a more classical design approach.

Regarding the SS a different approach was followed using radiation-tolerant components which are a cheaper solution – immunity to Single Event Upsets can be covered by other means. Also plastic packages are considered for radiative shielding since they are cheaper and more compact than ceramic packages. This low cost approach was deemed acceptable due to the short mission of the smallsats at the target (6 months) and the shielding provided by the MC during the transfer.

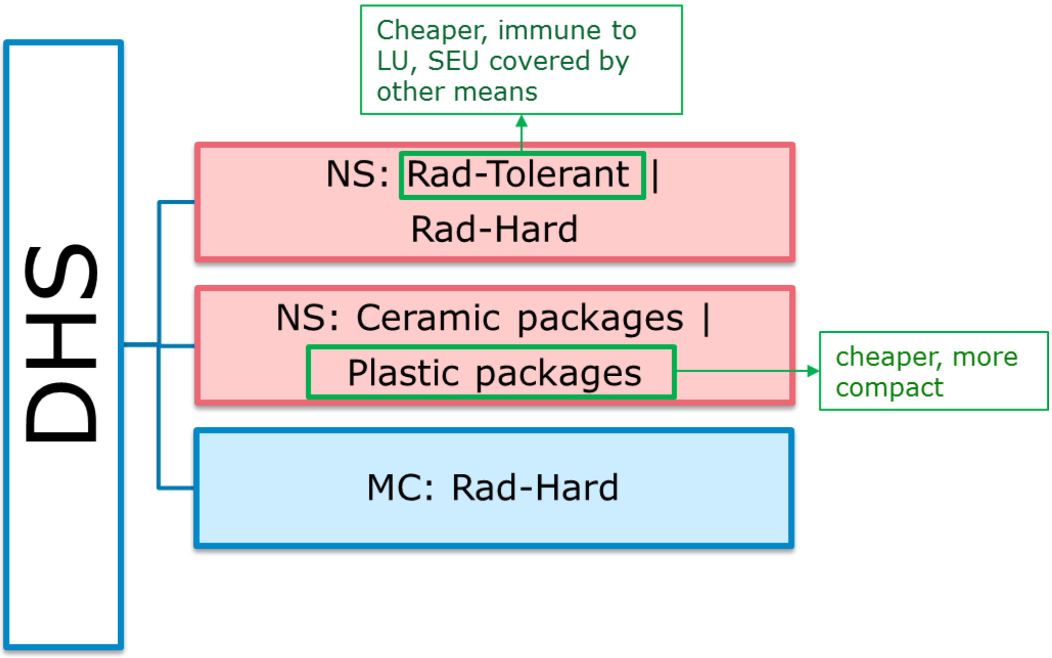


Figure 3-8: Graphical representation of DHS trade-off

3.6.5 GNC Trade-Off

For the MC the GNC actuation will be provided by Reaction Wheels and the gimbal mechanism on the main EP thruster and a cold gas system will be used to off load the RWs. On the sensor side it was decided to include a NAVCAM as it improves operations flexibility. The inclusion of this sensor may be traded-off later in the design.

For the SS, a combination of reaction wheels with a 3 DoF or a 6 DoF cold gas system was traded-off. The 6 DoF option was baselined to increase the operational flexibility. These operations may vary based on the target and a more robust solution is more adaptable to a wider range of targets. Furthermore, sun sensors, star tracker and IMU are foreseen. The use of an extra NAVCAM supported by an altimeter in the SS design was also baselined to improve operations and scientific return around the target. Regarding operations around the target, this is strongly dependant on the properties of the target, like size and knowledge of the gravitational field. For targets with diameters below 1 km hyperbolic arcs around the target were selected as a baseline. These need to be adapted depending on the target properties but this strategy still gives an advantage in operation costs and provides safer trajectories. More details on this trade-off are provided in 3.8.3.

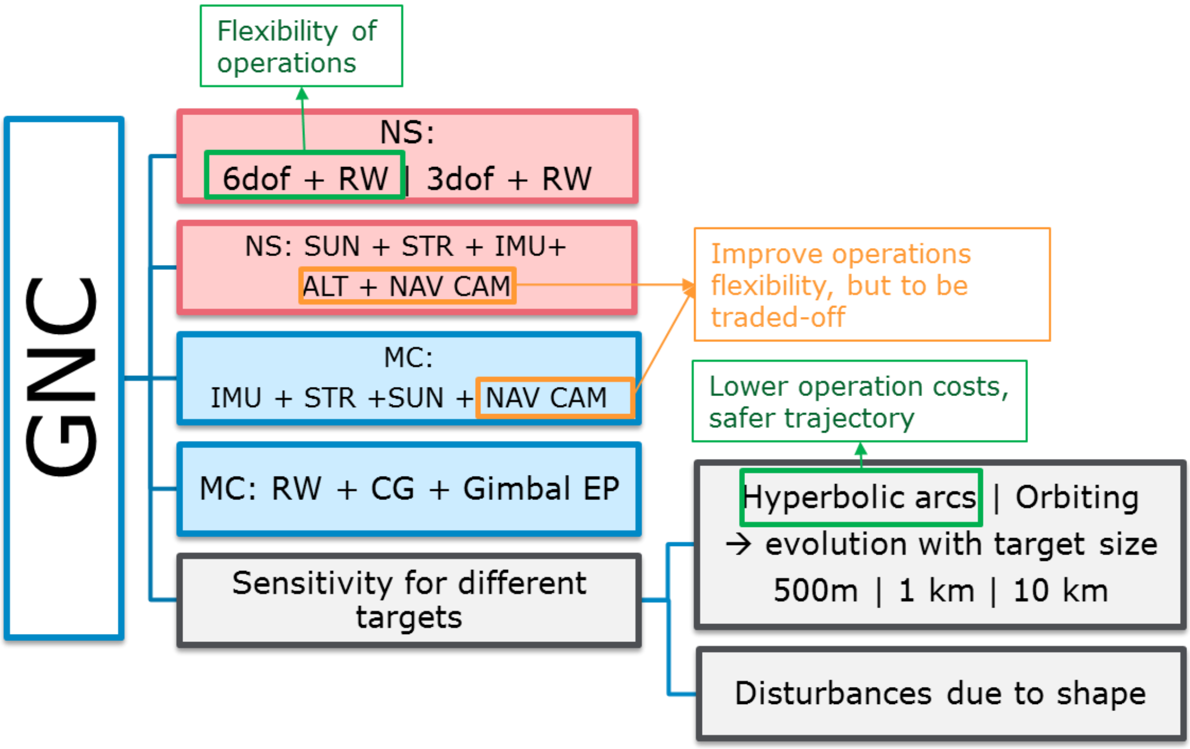


Figure 3-9: Graphical representation of GNC trade-off

3.6.6 Thermal Trade-Off

The Thermal Control System is highly dependent on the environment and the power it has to dissipate from the system. Hence the design is very dependent on the target orbital characteristics.

In order to maximise absorption, black MLI was selected for the SS in both mission scenarios to cope with the cold case. Golden MLI is more sensitive to variations in environment. Fixed radiator offers a simpler and lighter solution. For the Asteroid Belt mission, there is no need to include radiators as with the relatively low power dissipation the leaks of the MLI are enough to keep the temperature within the required limits. In the mission to NEO if the power resources are maximised to accommodate high payload needs, deployable radiators on the SS will be needed for maximum dissipation and allow for an adaptable area to fit the changing dissipation and environmental requirements. SS attitude restrictions – based on the radiator configuration - while pointing to the target would have to be defined.

On the MC, the black MLI was also selected and fixed radiators allow for the same benefits indicated for the SS. However to cope with the different environments at Earth and at the target louvres and loop heat pipes are used to minimise the dissipation at target when the Electric Propulsion is not being used and the solar flux is minimal (cold case), in particular for the MBA mission.

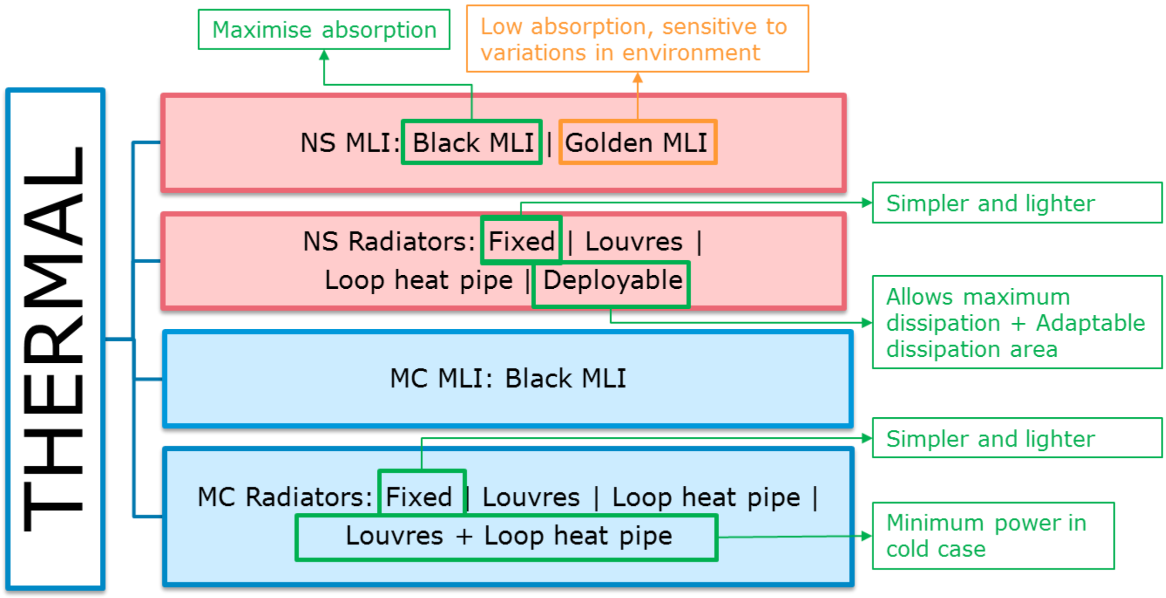


Figure 3-10: Graphical representation of Thermal trade-off

3.6.7 Operations Trade-Off

Mission Operations of the MC will commence at separation of the satellite from the launcher and will continue until the end of the mission, when the ground contact to the spacecraft will be aborted.

All ground communications with MC are via X-Band.

A 3 year period is assumed for mission preparation (as per AIM-Next).

No additional simulation campaign is foreseen in preparation of the operations at the target body, due to the limited GS&Ops support envisaged for this mission.

The Deep Station allocation will be decided once there is a final target selection and it will be based on the ground station coverage performed by Mission Analysis. The LEOP ground station coverage will be quasi-continuous and will have to be analysed once the final launcher is assigned, and it will need to consider the co-passenger strategies if the launch is shared with another mission. Non-contact periods for "passive" cruise should be in the order of 7 (EP)-14 days; anything above/below is likely to cause major impacts on the spacecraft ground segment design.

There are no stringent navigation requirements and minimum or no planning tasks. There is a potential delta-V saving for the MBA (~1 km/s) with Mars gravity assisted manoeuvre and it would allow for a wider range of target inclinations.

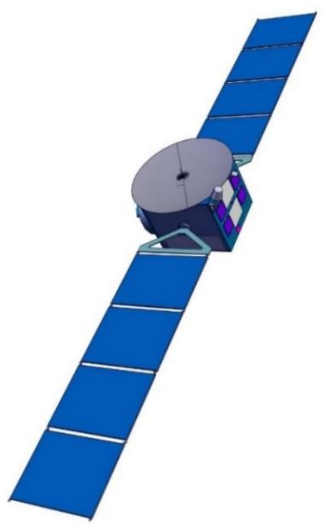
The SS Operations Concept is similar to the MC, with details provided in dedicated chapter of the NEO Report and MAB Report

3.7 Baseline Designs

Regarding the smallsats design, it is important to clarify that while a dedicated design would provide flexibility in the payload accommodation and allow for payload protrusions, for the purpose of this study it was decided to stick to the cubesat form factors in order to analyse the reusability of existing technologies and take advantage of the already existing cubesat deployers and available interfaces. However, this choice would have to be carefully reassessed in future steps since a tailor design for the smallsat offers clear advantages and flexibility.

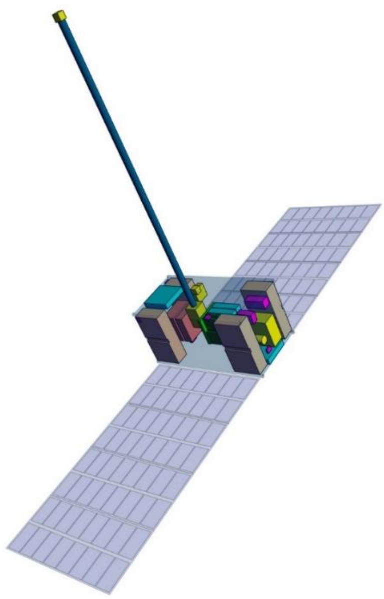
3.7.1 Option 1 - NEO Inactive Bodies

The baseline mission characteristics are outlined in Table 3-11 and Table 3-12.

Mother Spacecraft		
Dimensions (m)	2.0 x 2.0 x 2.2	
Dry Mass incl. margin (kg)	554.48	
Wet Mass incl. margin (kg)	784.36	
Power available to Electric Propulsion System at 1.1 AU (kW)	1.9	
Thrust level at 1.1 AU (mN)	84	
Specific Impulse at 1.1 AU (s)	1640	
Delta-V (m/s)	4530 for the transfer (2 years) 10 at target + RW desaturation	
Payload	-	
AOGNC	Sensors: IMU | STR | SUN | NAV CAM	
	Actuators: RW | CG | Gimbal EP	
Communications	Earth link: X band 2m HGA - 8h of contact with Ground Station	

Mother Spacecraft	
	ISL: 2 S-band LGAs
Data handling	OBC: Rad-hard components
Mechanisms	SADM | EP Gimbal | 4 Smallsats deployer
Electric Propulsion	2 propellant tanks by Orbital ATK of each 135 kg Xe storage capability, 1 high pressure regulator
	2 HET PPS thrusters (variable thrust and ISP), 1 thruster pointing mechanism, 2 Xenon flow controllers, 2 PPU, 2 EFU, 1 Pressure Regulation Electronic Card
	1 Cold Thruster assembly
Power	2 solar arrays with a total area of 8.3 m² with power generation optimised by SADM (MEC)
	20 kg PCDU and 10.26 kg battery (ABSL manufacture)
Structures	81kg
Thermal	Radiators - 0.83 m²
	Kapton Multi Layered Insulation, loop heat pipes

Table 3-11 MC Design Summary

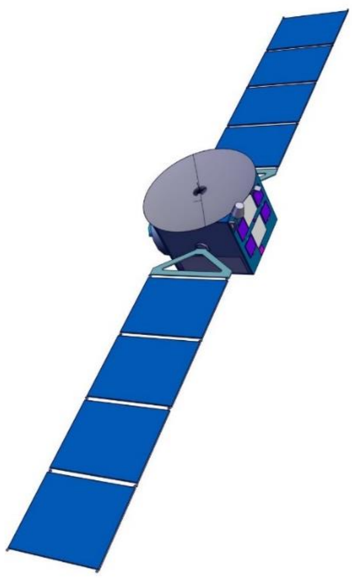
Smallsat (x4)		
Dimensions (m)	0.26 x 0.23 x 0.45	
Dry Mass incl. margin (kg)	28.87	
Wet Mass incl. margin (kg)	29.04	
Power generation at 1.1 AU (W)	117	
Delta-V (m/s)	10 at target	
Payload	Low frequency radar High frequency radar Camera IR spectrometer 159 Gbit expected data return	
AOGNC	Sensors: IMU | STR | SUN | NAV CAM	
	Actuators: RW | CG | Gimbal EP	
Communications	ISL: 2 S-band LGAs	

Smallsat (x4)	
Data handling	OBC: Rad-tolerant components
Mechanisms	SADM
Chemical Propulsion	Butane Cold gas system ~520 g Cold gas system
Power	2 solar arrays with a total area of 0.64 m² with power generation optimised by SADM (MEC)
	0.86 kg battery
Structures	16U SmallSat of the shelf Structure 2.25kg
Thermal	Black MLI chosen to maximize absorption at the target
	Radiators 0.33 m² – deployable radiators needed

Table 3-12: SS Design Summary

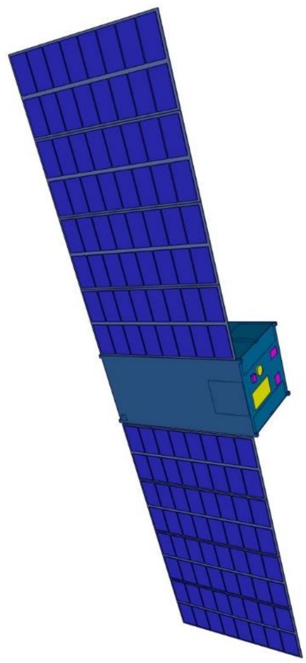
3.7.2 Option 2 - Main Asteroid Belt Active Bodies

The baseline mission characteristics are outlined in Table 3-13 and Table 3-14.

Mother Spacecraft		
Dimensions (m)	2.0 x 2.0 x 2.2	
Dry Mass incl. margin (kg)	747.48	
Wet Mass incl. margin (kg)	996.05	
Power available to Electric Propulsion System at 2.5 AU (kW)	1.5	
Thrust level at 2.5 AU (mN)	145	
Specific Impulse at 2.5 AU (s)	3540	
Delta-V (m/s)	11000 for the transfer (4 years) 10 at target + RW desaturation	
Payload	-	
AOGNC	Sensors: IMU | STR | SUN | NAV CAM	
	Actuators: RW | CG | Gimbal EP	
Communications	Earth link: X band 2m HGA - 16h of contact with Ground Station	

Mother Spacecraft	
	ISL: 2 S-band LGAs
Data handling	OBC: Rad-hard components
Mechanisms	SADM | EP Gimbal | 4 Smallsats deployer
Electric Propulsion	1 propellant tank capable of containing up to kg of Xenon, 1 high pressure regulator2 propellant tanks by Orbital ATK of each 135 kg Xe storage capability, 1 high pressure regulator
	Redundant T6 system, 1 thruster pointing mechanism, 2 Xenon flow controllers, 2 PPU, 2 EFU, 1 Pressure Regulation Electronic Card
	1 Cold Thruster assembly
Power	2 solar arrays with a total area of 26 m² with power generation optimised by SADM (MEC)
	PCDU and 12 kg battery (ABSL manufacture)
Structures	81kg
Thermal	Radiators – 2.35 m²
	Black Multi Layered Insulation, louver + loop heat pipes

Table 3-13: MC Design Summary

Smallsat (x4)		
Dimensions (m)	0.26 x 0.23 x 0.45	
Dry Mass incl. margin (kg)	22.33	
Wet Mass incl. margin (kg)	22.86	
Power generation at 2.5 AU (W)	28	
Delta-V (m/s)	10 at target	
Payload	Mass spectrometer Pressure sensor Ion/neutral spectrometer Magnetometer Camera Ion/electron spectrometer IR spectrometer 73 Gbit expected data return	
AOGNC	Sensors: IMU | STR | SUN | NAV CAM	
	Actuators: RW | CG | Gimbal EP	
Communications	ISL: 2 S-band LGAs	

Smallsat (x4)	
Data handling	OBC: Rad-tolerant components
Mechanisms	SADM
Chemical Propulsion	Butane Cold gas system (~520 g)
Power	2 solar arrays with a total area of 0.64 m² with power generation optimised by SADM
	0.49 kg battery
Structures	16U CubeSat of the shelf Structure 2.25kg
Thermal	Black MLI chosen to maximise absorption at the target
	No radiators

Table 3-14: SS Design Summary

3.8 Sensitivity Analysis

3.8.1 Launch to LEO with Epsilon/VEGA

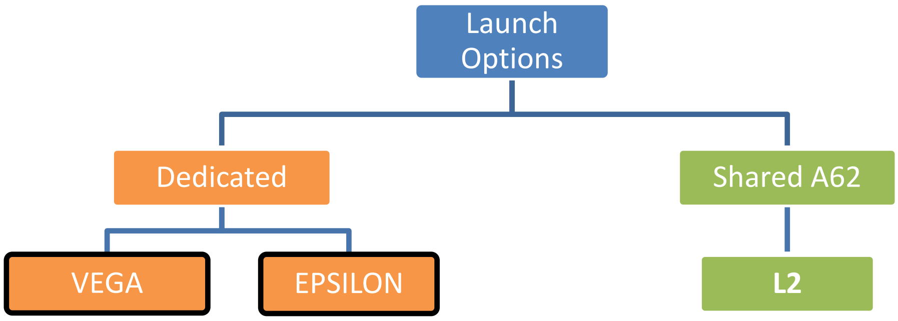


Figure 3-11: Launch options with baseline in green and sensitivity options in orange

The possibility of launching with a small dedicated launcher such as Epsilon/VEGA/VEGA C for the mission targeting a NEO was also assessed.

The T6 redundant EP system used for option 2 was baselined in order to optimise the wet mass of the S/C. The challenging case of 900 kg launch mass to a 200x4500 km orbit was assumed (optimised for this launch mass and to spend less time in orbits crossing the inner Van Allen belt). The assumptions for this analysis are in line with the ones reported in section 3.4. To calculate the orbit raising time the thrust level is 145 mN with 5 kW available at 1 AU, the ISP is at its best at 4048 s, and a duty cycle of 90% (10% required to account for NAV, comms, EP outages and contingencies), a 10% margin to account for eclipse time and 15 days for commissioning in LEO.

With the T6, the escape brings us to a total propellant mass of 132 kg. The time to escape including the eclipse margin and duty cycle mentioned above is of 530 days and 230+ days spent in the Van Allen belt was considered as the driver for radiation assessment.

	T6	PPS1350
Escape Delta V (m/s)	~6300	
Propellant mass (kg)	132	292
Time to escape (days)	~530	~805
Time to be above the inner Van Allen belt (days)	~230	~365

Table 3-15: Launch to LEO with Epsilon – EP options

To look at the complete mission, the transfer from Earth escape to target with T6 with the following envelope values was added:

- Delta-v: 3998 m/s
- Propellant mass: 87 kg
- Departure date: 2027/04/29
- Total transfer time: ~2 year

	T6
Delta V (m/s)	~11300
Propellant mass (kg)	~220 kg
Total duration (days)	~1265 = ~3.5 years

Table 3-16: Complete mission with T6

Mass wise this solution seems to be feasible, but there are still many open points.

Extra shielding may be needed to cross the Van Allen belt (230+ days): doses and proton induced single event effects would have to be considered. The dose will also be accumulated during the transfer through the outer electron belt.

The mass criticality can be compensated with a launch into a lower orbit with consequences on the duration of escape and on transfer duration - 1200 kg leads to a total duration of ~4.5 years.

The solar arrays would need to be designed to cope with degradation and the batteries would need to be resized to cope with eclipses.

Due to the dimensions of the fairing the accommodation of a HGA needs to be further assessed.

Finally, the higher longitudinal and lateral mode fundamental frequency requirements with Epsilon, ≥ 30 Hz and ≥ 10 Hz respectively, are going to have a significant impact on the needed structural stiffness of the spacecraft. Consequently, the Design Limit Load (DLL) of the instruments will increase with Epsilon.

3.8.2 SADM Option on the Smallsats

The option of not carrying a SADM on the SS was assessed.

In particular for targets in the Asteroid Belt the power generation is critical, and the SADM provides more operational flexibility.

To avoid the use of a SADM, it would be necessary to stay in a close to Dawn-Dusk orbit (permanently bathed in sunlight) which would compromise the scientific objectives of getting observations of the dark side of the target. The option of reducing the power consumption by reducing the duty cycles of the instruments would also compromise scientific objectives. Additionally, the batteries may also need to be resized to cope with peak power.

3.8.3 Sensitivity to Target Size

The possibility of selecting different target sizes was evaluated at system level.

The manoeuvre selected for AOCS of hyperbolic arcs is safer and less sensitive to knowledge of the gravity field which minimises operations complexity and cost and allows for safe mode, reducing the risk of collision. However, this approach is quite sensitive to the target's size and requires a minimum distance to the target. The delta-V required for these hyperbolic arcs for a given minimum distance increases with the size of the target in a ratio of $\sqrt{R^3}$. A larger distance to the target could also be envisaged but it will imply a degradation of the science objectives.

For larger targets (> 1 km diameter) it is necessary to include insertion into a stable orbit or to implement the necessary design changes to allow for a higher delta-V capability. In this scenario, the mother spacecraft visibility of all the smallsats is more limited. A mesh communication architecture could be an enabler for this mission setting.

The proposed strategy for targets with diameters of 1 km and above is for the mother spacecraft to stay in hyperbolic arcs with an increased distance to the target, which will have an impact on the ISL, or to perform more frequent manoeuvres, e.g. every 3-4 days. The smallsats could be inserted into a 5 km altitude stable orbit around the target, which implies higher operations complexity, and requires more accurate knowledge of the gravity field. Therefore, the insertion should be done in a stepped approach, starting from high hyperbolas to gain more knowledge of the target and only then approach the target and proceed to the insertion into the final orbit. The Safe Mode and FDIR impacts must be analysed. Optionally, it would also be possible to increase the distance to the target but this would compromise scientific requirements.

3.8.4 Asteroid Impactor or Lander

Considering the case in which one of the smallsats fits the role of an asteroid impactor or lander, the design of this SS is significantly impacted. In particular the requirements become more stringent, in particular for a fast spinning asteroid due to the increased complexity for AOGNC and operations.

The easiest option with smaller impact on delta-V is to have an impactor. The visibility from the mother spacecraft is limited during descent and for scientific data download. A Mesh communication architecture could be an enabler.

To have a lander being delivered by the mother spacecraft, the smallsats would have to be more autonomous which would increase the design complexity. (Reference to FASTMOPS study which covers the lander delivery timeline and requirements).

3.8.5 Martian Moon Target – Phobos

The case of Phobos as a target was only assessed qualitatively in order to carry out an initial evaluation of the requirements and main impacts on the mission design.

The transfer to Phobos would include Mars injection and spiralling down manoeuvres – for 2028, this would translate into a delta-V requirement of 8.5 km/s. The mass of the target is significantly higher than what was assessed during the study for asteroids and also has the additional factor of being in Mars proximity. Consequently, the hyperbola hopping for ~5 km minimum altitude is not feasible and the the "low cost" operational concept is no longer applicable. Additionally, the distance between the mother spacecraft and the smallsat will need to increase.

Depending on the season, it might be needed to account for Mars eclipses lasting up to 55 min and 2 to 3 extra hours.

In order to navigate to Mars, a Delta DOR system and known ephemeris as well as available relay orbiters could be used, and would be an advantage.

The mother spacecraft can enter a Quasi-Satellite Orbit (QSO) with Phobos, which will result in an increased delta-V requirement for station keeping manoeuvres and a significantly increased distance to the SS. To provide hundreds m/s, the propulsion options feasibility will need to be reassessed. A hydrazine system could be necessary.

Additionally, a MGA would be needed to cope with the increased distance between the mother spacecraft and the smallsats. During the study, navigation based on *line of sight* was considered, in this case, limb detection may be needed (wider angle camera) and a higher DHS processing power would be required.

For the smallsats to enter an orbit around Phobos, more delta-V is required (e.g. for 100 x 50 km altitude - ~10 m/s order delta V per manoeuvre / every 5 days), and better knowledge of gravitational field and landmarks is needed. Additionally, a larger FoV Camera would be required and the smallsats would have to be more autonomous. Because of the larger DV requirement imposed on the smallsat by this option, it would probably make more sense to equip the smallsats with an electric propulsion thruster in case Phobos would be the selected target of study.

An alternative strategy is to consider regular fly-bys of Phobos. At every fly-by one smallsat could be released and be left drifting to get the science data while being sufficiently close, which is the main advantage of having the smallsats. This would require high separation velocity.

3.9 Technology Requirements

The following technologies are required or would be beneficial to this domain:

Included in this table are:

- Technologies to be (further) developed
- Technologies available within European non-space sector(s)

- Technologies identified as coming from outside ESA member states.

Sub - System	Equipment and Text Reference	Technology	Suppliers and TRL Level	Technology from Non-Space Sectors	Additional Information
Mechanisms	LV-POD	Low velocity CubeSat Deployer	ISIS (NL) TRL 6		The low velocity technology is expected to be used in flight on the RemoveDEBRIS mission, however this will require modification to accommodate the larger SS for the SPP Mission
Mechanisms	SAC	SS SADM	IMT (I) TRL 3		An activity is on going to increase the TRL level to 6, however this may require modification to accommodate and equivalent of 4 panels on a 2U wide platform as these are not strict requirements for the development.
Chemical Propulsion	Kick-stage applications	Water propulsion for kick-stage applications	-	NO	This technology would be beneficial in terms of kick-stage application. Since the corresponding system have the potential of increasing the overall specific impulse the performance of the kick-stage can be improved.
Chemical Propulsion	High Performance Cubesat Propulsion System	e.g. Mono-/Bipropellant System	See [REF CPROP Table 1.1]	NO	
Chemical Propulsion	Deep Space Qualification for Cubesat Propulsion Systems	-	-	-	
Electric Propulsion	PPS1350-E	Hall effect thruster	Safran-Snecma	NO	Ongoing qualification for stationkeeping purposes. To be assessed whether delta qualification required for transfer to NEO
Electric Propulsion	PPU Mk2	PPU	TAS Belgium	NO	To be assessed whether delta qualification required

Electric Propulsion	EGEP PSCU	PPU	Airbus CRISA / Airbus Friedrichshafen	NO	The Equipment is under development under EGEP targeting TRL 5. Qualification shall be performed. Further, capability for beam voltage variation could be implemented to increase performance with varying power input.
Electric Propulsion	T6 & FCU	GIE & FCU	QinetiQ	NO	TBC if delta qualification would be needed. Lessons learnt from the BepiColombo flight qualification tests shall be used to improve the GIE design and performance. Tuneable beam voltage and grid optimization are to be investigated to enhance performance as a function of varying input power. An increase in specific impulse could be achieved by implementing a 4-grid concept (low TRL).
Electric Propulsion	Xenon tanks	Tank	MT Aerospace	NO	Potential European supplier; preliminary design exists
Electric Propulsion	HPR & FCU	Propellant management	AST / Nanospace	NO	Low-mass developments alternative to baseline equipment
Electric Propulsion	T5 (Option)	GIE	QinetiQ	NO	Higher beam voltage to be implemented (delta qualification required)
Electric Propulsion	T5 Gimbal (Option)	Thrust Vector Control	RUAG Space Austria	NO	No COTS gimbal for the T5 exists, but a delta design from existing gimbals could be considered.
Electric Propulsion	PPTCUP (Option)	PPT	MarsSpace & ClydeSpace	NO	Delta qual/design required for radiation toughness/hardness for deep-space operation

Electric Propulsion	Electrospray thruster (Option)	Colloidal thruster	Queen Mary University	NO	Currently under EPIC funding to bring to TRL 5
GNC	[REF GNC Section 1.5]	Semi-autonomous attitude guidance based on LOS navigation in asteroids	ADS, GMV (TRL-4)	N/A	Activity pre-development for AIM
GNC	[REF GNC Section 1.11]	Semi-autonomous attitude guidance based on LOS navigation in Phobos mission	ADS, GMV (TRL-4)	N/A	Limb-detection for spherical bodies implemented in JUICE
GNC	[REF GNC Section 1.12]	GNC for asteroid landing	ADS , GMV (TRL-5)	N/A	Developments carried out for MarcoPolo and MarcoPolo-R
Thermal	Smallsat	Deployable Radiator	TRL3 in Europe TRL6 for US companies	-	GSTP initiated
Thermal	Mothercraft	Louvered Radiator	TRL6 SENER at least delta-qualification, but potentially re-design necessary.	-	TRL9 for SENER louvers for ROSETTA.
Operations					There are ground technologies beneficial to the Ground Segment and Operations for operating SPP missions. All those that will improve and reduce the limitations imposed by cost constraints and that will not add additional work to the overall mission design.

Table 3-17: Technology Development overview by subsystems (blue: Targets in the Main Belt; green: NEO like Targets; white: applicable for NEO like and Main Belt Targets)

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4 PROGRAMMATICS/AIV - ALL OPTIONS

4.1 Requirements and Design Drivers

Only one specific driving requirement for AIV/programmatics has been defined for this study. It relates to the schedule and concerns the launch date to be between 2024 and 2034 [MIS-070].

Another driver, although not directly stipulated in a requirement, is the use of Ariane 6 as the preferred launch vehicle. Ariane 6 is currently being designed and has its first test flight scheduled for 2020.

4.2 Technology Readiness Levels & Technology Developments

The Technology Readiness Levels (TRL) present a systematic measure, supporting the assessments of the maturity of a technology of interest and enabling a consistent comparison in terms of development status between different technologies.

The TRL definitions from RD[2] are shown in Table 4-1:

TRL	ISO Definition	Associated Model
1	Basic principles observed and reported	Not applicable
2	Technology concept and/or application formulated	Not applicable
3	Analytical and experimental critical function and/or characteristic proof-of concept	Mathematical models, supported e.g. by sample tests
4	Component and/or breadboard validation in laboratory environment	Breadboard
5	Component and/or breadboard critical function verification in a relevant environment	Scaled EM for the critical functions
6	Model demonstrating the critical functions of the element in a relevant environment	Full scale EM(s), SM, STM, TM, DM(s),representative for critical functions
7	Model demonstrating the element performance for the operational environment	QM
8	Actual system completed and “flight qualified” through test and demonstration	FM acceptance tested, integrated in the final system
9	Actual system “flight proven” through successful mission operations	FM, flight proven

Table 4-1: TRL scale

A general statement can be made that only technology sufficiently advanced (i.e. to TRL6) can be considered to be mature enough to be included at the start of the

Implementation Phase. Since there are low TRLs between 3 and 5 identified within the study, predevelopment activities will need to be performed in order to raise the respective subsystems to TRL 6. The developments are discussed in more details in the corresponding subsystem main chapters, whereas their programmatic impact is discussed in the following subsections 4.2.1, 4.2.2, and 4.5.4.

Table 4-2 shows a general indication of the development times depending on the current TRL. According to the European Space Technology Master Plan, when preparing the contractual basis for multi-annual programs, it takes about 18 months to reach political agreement on financial ceiling. This has also been included in the table.

TRL	Duration
5-6	4 years + 1.5 year
4-5	6 years + 1.5 year
3-4	8 years + 1.5 year
2-3	10 years + 1.5 year
1-2	12 years + 1.5 year

Table 4-2: TRL – development duration

Assuming, that the development of technology at a TRL lower than 6 is already approved and on-going, we can expect that we need another 2 years before the implementation phase can start for technologies at TRL 4 and another 4 years for technologies at TRL 3 unless very special effort is made to speed up the development.

The purpose of the above table is to give the reader a general indication on the development times to be expected. These times can of course vary depending on the technical subsystem and its specific required development activities. In case of the SPP study, a first assessment has been performed for all low TRL technologies with respect to the technical time required to mature them to TRL6. These can be found in subsections 4.2.1 and 4.2.2.

4.2.1 Technology Readiness Levels MC

The product tree for the MC is shown in Table 4-3. It identifies for each subsystem the associated equipment, sometimes components, their TRL as far as available, and also lists heritage reference(s) (i.e. which missions has it or will it be flown on).

The table is valid both for Option 1 and Option 2. The only difference between the two with respect to technology used is the electric propulsion subsystem. Both options are listed in the table.

Subsystem	Equipment	TRL	Reference
AOGNC	Sun Sensor	9	ExoMars
	RW	9	Proba 2

Subsystem	Equipment	TRL	Reference
	NAVCAM	6	ISS, Prisma
	STR+IMU	9	Proba 3
COMMS	TWTA	6	
	X-Band Deep space Transponder	6	Juice, ExoMars, Bepi Colombo
	2m antenna	6	Juice
	ISL	5	Proba 3
EPROP for Option 2	T6	7	BepiColombo
	TPA Gimbal	7	BepiColombo
	PPU	6	BepiColombo, G2G
	FCU	7	BepiColombo
	PSA	8	SmallGEO
	Electronics	5	BepiColombo
	Tank	9	AIM Next
	Xe Cold Gas Thruster	7	Swarm, Tandem-X (w/ other gases)
EPROP for Option 1	PPS1350-E	6	SMART-1, AlphaSat
	TPM Gimbal	8	Telecom satellites
	PPU	7	Telecom satellites
	Tank	8	AEHF
	XFC	9	Telecom satellites
	Electronics	5	SMART-1
	Xe Cold Gas Thruster	7	Swarm, Tandem-X (w/ other gases)
DHS	OBC Backplane	6	MASCOT 1 Lander, MASCOT 2
	OBC MM RTU	6	MASCOT 1 Lander, MASCOT 2
MECH	SS Deployer	6	AIM (with delta development)
	SADM	9	Significant heritage in EO
PWR	Battery	9	GAIA or SWARM
	Solar Array	9	EDRS-C
	PCDU	7	BepiColombo MTM
STRU		6	
TCS	Heater	9	Extensive EO, NAV, TIA, SCI
	MLI	9	Extensive EO, NAV, TIA, SCI
	Standard radiator	9	Extensive EO, NAV, TIA, SCI
	Thermistor	9	Extensive EO, NAV, TIA, SCI

Subsystem	Equipment	TRL	Reference
	Heat Pipe	9	Extensive NAV, TIA,
	Thermal Filler	9	Extensive EO, NAV, TIA, SCI
	Thermal Paint	9	Extensive EO, NAV, TIA, SCI
	Louver	4	Rosetta; JUICE
	Loop Heat Pipe	9	Telecom satellites

Table 4-3: MC TRL levels and heritage references

Table 4-4 below lists components of subsystems with lower TRL or special aspects in conjunction with their technology status. They have been specifically highlighted and analysed for their overall impact on the schedule and development activities. All equipment mentioned in the table require either a pre-development to TRL6 or need to be kept under close monitoring if they are to be considered for SPP. In case a pre-development time has already been identified for the equipment, it is mentioned in the corresponding column.

Sub-system	Equipment	Estimated technical pre-development time to TRL 6 [yrs]	Remarks
AOGNC	NAVCAM	2	Flight heritage on ISS, Prisma (LEO environment); delta development and qualification for deep space environment required
COMM	Inter Satellite Link	3	To be flown on Proba 3 Adaptation to SPP mission parameters requires an estimated predevelopment time of 3 years
E-PROP	Electronics		Electronics need to be redeveloped, can be done within the nominal implementation phases (EM/QM/FM).
	Tank		Tank for Option 2 developed for AIM Next; status of AIM Next and subsequent TRL level has to be reassessed at mission definition. At present, a TRL of 6 with the need for a delta qualification is assumed. Tank for Option 1 from the US, ITAR restrictions may apply. European tank not yet available, but development within 2 years feasible.
	Thruster		Requires delta-qualification activities to SPP parameters

Sub-system	Equipment	Estimated technical pre-development time to TRL 6 [yrs]	Remarks
MECH	SS deployer		Development is foreseen for AIM; status of AIM and subsequent TRL level has to be reassessed at mission definition. At present, a TRL of 6 with the need for a delta qualification is assumed.
DHS	OBC Backplane		Heritage on lander MASCOT 1/2 Adaptation from lander to satellite requirements to be analysed for required delta developments
	OBC MM RTU		Heritage on lander MASCOT 1/2 Adaptation from lander to satellite requirements to be analysed for required delta developments
TCS	Louver	2	To be flown on Juice A pre-development time of 2 years to TRL 6 for SPP is expected.

Table 4-4: Overview technology pre-developments for MC

4.2.2 Technology Readiness Levels SS

The product tree for the SS is shown in Table 4-5. It identifies for each subsystem the associated equipment, sometimes components, and their TRL as far as available with corresponding technical development times for the technology to reach TRL6. It became clear during the study that many (though not all) components for the SS are based on components that have already flown on CubeSats or are in the process of being qualified for CubeSat use. All of these components are marked in the last column.

Subsytem	Component	TRL "classic"	Time to TRL6 [yrs]	TRL "Cubesat"	Time to TRL6 [yrs]	"Cubesat" baseline?
AOGNC	Sun Sensor	6		4	3	X
	Altimeter			4-5	2-4	X
	IMU			4-5	2	X
	NAVCAM			6	2	X
	RW			6	1	X
COMMS	Antenna	6		6		
	ISL	6	3	6		
CPROP	Thruster			5-6	2	X
	Tank			5-6	2	X
	EPROP - not baseline			(7)		(X)
DHS	Platform OBC			4	1	X
	Payload OBC			6		X
	Dock Board			6		X

Subsytem	Component	TRL "classic"	Time to TRL6 [yrs]	TRL "Cubesat"	Time to TRL6 [yrs]	"Cubesat" baseline?
MECH	SAC	3	2			
PWR	Battery					X
	Solar Array	6				
	PCDU					X
STRU						
TCS	Heater	9				
	Thermistor	9				
	MLI	9				
	Thermal Filler	9				
	Paint	9				
	Deployable Radiator	3-4	2			

Table 4-5: SS overview TRL levels

All systems that rely on a "CubeSat" baseline need to be (delta-) qualified for the target deep space mission environment. This also applies for technologies already at TRL6, as their standard target environment is LEO. In addition, they also have certain commonalities that significantly differ from ECSS based technology developments and which should be addressed in order to increase the success rate when performing a qualification campaign. These are detailed in section 4.3.

The TRLs listed in Table 4-5 for the "CubeSat" baseline equipment are all for the specific pieces of hardware chosen by the respective technical domains within this study. It has not been the objective of the SPP study to produce an extensive overview of existing CubeSat solutions, thus other pieces of equipment with comparable performances may exist having different TRL levels. The stated TRL levels in Table 4-5 are valid for the specific hardware in SPP, but should be considered only indicative for the equipment branch. A detailed assessment of the TRLs is presently necessary for every future technological mission scenario.

For all other subsystems not using CubeSat technology as a baseline, the following comments apply:

- COMM: ISL to be flown on Proba 3, the adaptation to SPP mission parameters requires an estimated predevelopment time of 3 years
- TCS:
  - Deployable Radiator has an expected technical pre-development time of 2 years
  - Uses EO,NAV,TIA,SCI heritage for all other components
- MECH uses M-ARGO concepts; an estimated technical pre-development time of 2 years is required for SPP
- PWR SA uses Proba-V and Cheops heritage.

It should also be highlighted that the baseline solution for the chemical propulsion is based on a cold gas thruster with a TRL between 5 and 6. However, a similarly performing option based on an electrical thruster (TRL7 - TRP T718-176MP) exists.

4.3 General Development Approach COTS Space Systems

A central aspect during the SPP study is the possible use of hardware like COTS and CubeSat solutions, specifically for the SS. As seen in section 4.2.2, many of the baselined subsystems for the SS already use this branch of technology for the SS. Most of these technologies can be considered "low cost" when comparing them to ECSS-based technology developments. The low cost is as much related to the actual buying price of components as to the costs (and times) associated with their development (for custom designs).

In order to enable the use of as well as benefit from a COTS approach, the strategy in the SPP study is twofold:

The first element is aimed at a specific, already identified target technology. In order to verify that the COTS subsystem or component is suitable for deep space missions, a qualification campaign with suitable requirements needs to be performed.

The second element is based on the observation that all of the (existing) COTS technologies have certain features in common. Analysing these allows to derive a set of recommendations that can help in defining strategies lowering the threshold for successful implementation of these technologies to deep space missions.

Four observations and their corresponding recommendations are listed here below:

4.3.1 Operational Environment

Almost all previous missions for CubeSats or similar technology have been developed for operations in LEO. Aside from differences in communication strategies, the main environmental difference is the increased level of radiation a spacecraft is exposed to. For (deep space) missions outside of Earth's magnetic field, the spacecraft will be exposed to higher levels of fluxes due to a larger number of different particles, particle species and respective energy levels.

Recommendation:

- Perform radiation testing on candidate equipment at higher dosage levels and with different species (p+, n, e-, heavy ions). In general, these tests are considered feasible for CubeSat sized equipment.
- Define set of "CubeSat radiation mitigation design rules" for COTS systems
- Assess existing designs with regard to their compliance to ECSS.

4.3.2 Limited Mission Durations & Storage Lifetime

Mission durations of CubeSats can vary significantly, from a few days to a few months and even a few years. However, they are not systematically designed for longer mission durations, but rather on a case by case basis. Factors like choice of components and materials, system design, handling of components, component quality control, etc. can all have an influence on the maximum period of inactivity after which a system is still able function. In the case of SPP and deep space Science missions, hibernation and storage periods can be expected to be in the order of years. For most CubeSat systems, the storage lifetime is not systematically known.

Recommendation:

- Assess existing COTS/CubeSat systems for storage lifetimes
- Define a set of specifications for testing of storage lifetime.

4.3.3 Use of COTS Components from Proven and New Supply Chains

The origin of parts and components and their history is not always known. Depending on the supplier or manufacturer, the components may be from well controlled sources who adhere to agreed quality control procedures and processes, or they may be from suppliers whose quality control is not transparent to the customer or does not meet minimum requirements. Uncertain supply chains can lead to an increase in failure rates of components, which may or may not be detectable during testing.

Recommendation:

- Assess methods of how to ensure a certain quality of used components for DS SmallSats (e.g. trusted suppliers, whitelist DML, etc.)
- Evaluate scenarios for "reliability by testing", i.e. identify testing methods in addition to the QM/FM approach that can increase statistical trust in workmanship and sufficient quality of components (e.g. statistical batch testing).

4.3.4 Quick Development Time and Permitted Risk Attitude

CubeSats are developed with a different risk scenario in mind than ECSS based space missions. For CubeSats, "failure can be an option". This is often reflected in the designs and in the design approach. It usually permits a much quicker development time when compared to conventional systems, while leading to an increased risk for component performance and reliability.

Any standard application of ECSS would decrease this risk, while significantly increasing cost and development time, thereby severely restricting one of the big advantages of the CubeSat design approach. Therefore, a dedicated tailoring of the ECSS standards for CubeSats "in LEO" has already been created (i.e. the tailoring is based on a risk profile applicable to the operational scenarios and business cases of CubeSats flying in LEO). The risk profile for CubeSats in LEO is however not considered to be the same as the risk profile for deep space Science missions.

Recommendation:

- Re-evaluate the existing tailoring to ECSS for LEO Cubesats in order to adapt risk vs. time/cost to better fit the risk requirements for Science deep space missions.

4.4 Model Philosophy

Due to the significant differences in heritage of MC and SS, the model approach differs for both.

4.4.1 Model Philosophy MC

Analysing the heritage and TRLs of the MC, it is apparent that almost all proposed systems have some form of flight heritage. Most subsystems can therefore be considered to have TRL 6 or even higher.

For specific subsystems, due to the low level of heritage and/or TRL, the use of qualification models is foreseen.

Overall, the approach for the model philosophy of the MC is summarised in Table 4-6:

Models	**Remarks**
Engineering Model(s)	Equipment level, at various levels of complexity Models are to be used for ATB
QM	Equipment level, for: - NAVCAM - Electric Propulsion - SS Deployment Mechanism
SVF & ATB	Use of EM's and QMs for ATB
S(T)M	System level Decision on use of a thermal model needs to be taken in phase B. One aspect to be considered is system performance verification of TCS louvers, which presently have a TRL of 4.
PFM	System level

Table 4-6: Model philosophy for MC

4.4.2 Model Philosophy SS

Aside from components for the thermal subsystem and the solar array, almost all other subsystems are either conceptual or are based on CubeSat designs. The CubeSat designs have been developed for different business and usage cases resulting in varying levels of maturity. This low level of relevant heritage and maturity of most subsystems requires therefore either newly designed subsystems or delta-qualifications for adaptation to the project. A QM/FM approach is therefore used [RD[3]].

The overall model approach for a (first) SS is shown in Table 4-7.

Models	**Remarks**
Engineering Model(s)	Equipment level, at various levels of complexity Models are to be used for ATB
Flatbed (SVF & ATB)	Use of EM's
QM	Equipment level and system level
FM	System level

Table 4-7: Model philosophy SS first "batch"

Due to the nature of potential missions, more than one spacecraft could be used (SPP study includes four SS, though many more could be envisaged). These spacecraft may be completely different, identical, or similar (e.g. modified with different payloads):

- For different SS spacecraft, Table 4-7 is again applicable
- Identical (recurring) SS spacecraft will simply be produced as FMs
- Similar SS spacecraft should be produced as PFMs. The detailed model philosophy then takes into account the differences. E.g. for a significant difference in thermal or structural requirements, an STM will be built; new subsystems will be included via an QM/FM approach on equipment level (see Table 4-8).

Models	**Remarks**
Engineering Model(s)	Equipment level, at various levels of complexity Models are to be used for ATB
QM	Equipment level, if subsystem is new or requires delta-qualification
Flatbed (SVF & ATB)	Use of EM's
STM	Depending on differences
PFM	System level

Table 4-8: Model philosophy consecutive, similar SS spacecraft

4.4.3 Spare Philosophy

SPP is launch window driven, therefore a proper spare philosophy shall be implemented for MC as well as SS. However, a spare philosophy cannot be defined at this early stage in a project, since it depends on risk and reliability assessments, chosen components, budget constraints, model philosophy, to name a few. A definite approach describing the spare philosophy will need to be available at the PDR.

4.4.4 Test Facilities

No special limitations or constraints for test facilities have been identified for SPP.

The SS subsystems are small enough to be tested in small sized test facilities, while the MC can be tested in small to standard sized test facilities.

One aspect to highlight is that additional radiation tests are recommended for the COTS components for the SS. Due to their small size most of them can be tested in existing radiation chambers without major modifications.

4.5 Schedule

Aside from the assumed durations for the various activities, the sensitivity of the schedule for SPP is only driven by TRL levels, model philosophy, and procurement approaches. Since the technology development requirements between the different options do not differ much, and the procurement approach and model philosophies at this stage of the study are assumed to be similar between the options, only one schedule is used for the assessment of all options.

The schedule is based on a perspective of what could be if the mission was kicked off today. No specific key milestone and decision dates are incorporated, therefore any starting date for Phase A after 01.01.2018 simply moves the launch date the same amount of months into the future.

Two schedules are evaluated: a baseline a schedule and a schedule investigating an optimised approach for the SS implementation phase. It is a variation of the baseline schedule taking into account potential time savings due to the implementation of some of the synergies and benefits of a CubeSat development approach.

4.5.1 Assumptions

The following assumptions are used when drafting the baseline schedule. Any changes and additional assumptions for the SS optimised schedule are mentioned in the corresponding chapter 4.5.3.

	Assumptions
1	MDR/beginning of phase A on 01.01.2018 (“today”)
2	Review durations 30days
3	ITT 6 months each at start of Phase A/B1 and Phase B2/C/D
4	Phase durations baseline: • Phase A/B1 9 months • Phase B2 5 months • Phase C 12(SS) / 14(MC) months • Launch campaign 3 months • Contingency 3 months
5	Subsystem model durations for baseline (incl. equipment level testing): • EM 1-3 years • SS QM 1 year • SS FM 0.7 years • S(T)M 1.2 years • PFM 1-2 years
6	EM’s can be started in Phase C
7	Subsystem CDRs 3 (SS) to 4 (MC) months before system CDR
8	SS FM development starting 2 months before QR
9	Procurement of critical parts starting before CDR
10	ITAR: Tank and IMU procurement could start 1 year earlier (US supplier) -> decision point after PDR
11	SS not available for MC environmental test, joined “delta” environmental tests: • mainly vibration, mass properties, EMC • TVAC test not assumed
12	ECSS standard approach for MC and SS
13	Ariane6 launcher requirements known by 2020

4.5.2 Baseline Schedule

The baseline schedule (Figure 4-1 & Figure 4-2) results in the following major milestones:

- MDR (KO phase A) T0
- PDR in T0 + 26 months

CDR in May (SS) / T0 + 42 months (MC)

- Delivery of MC PFM T0 + 66 months
- QR for SS in T0 +65 months
- Delivery of SS FM in T0 + 80 months
- Launch date in T0 + 91 months

For the purpose of this study, the MC and the SS are treated as spacecraft that are designed and developed simultaneously up to the PDR. Detailed design, procurement and verification (Phase C/D) then are performed in parallel while allowing for physical and contractual separation of the activities.

The analysis of the schedule for the MC (Figure 4-1) shows that the critical path for the MC in Phase C and D is driven by three main factors:

- The electric propulsion development activities
- The S(T)M development and test campaign
- The NAVCAM QM/FM approach.

Each of these have a similar duration, so that in order to decrease the critical path for the MC, alternative approaches for all three of them at the same time would have to be found.

These three drivers allow a margin of 4-6 months for all other subsystems in Phase C/D.

For the development of the ground station, a time window of 5.5 years is available between PDR and launch date.

In the frame of the CDF Study T0 was assumed as 1 January 2018.

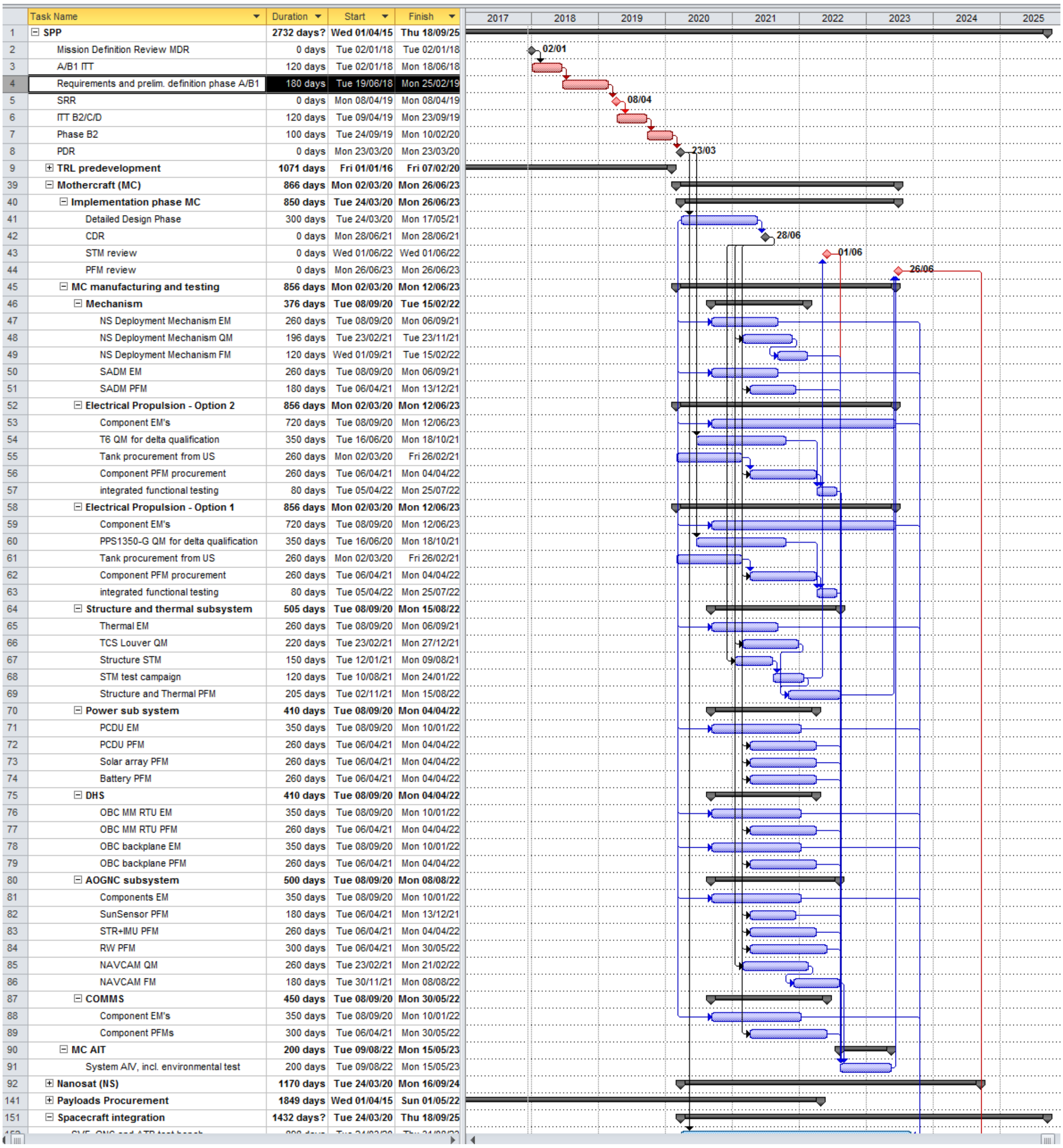

Figure 4-1: Baseline Schedule: Phase A/B1/B2 and MC

The baseline schedule for the SS and the launch date are shown in Figure 4-2. The critical path for the SS is driven by the chemical propulsion system. This results in a margin of approximately 4 months for other subsystems for the QM, and 2 months for the FM.

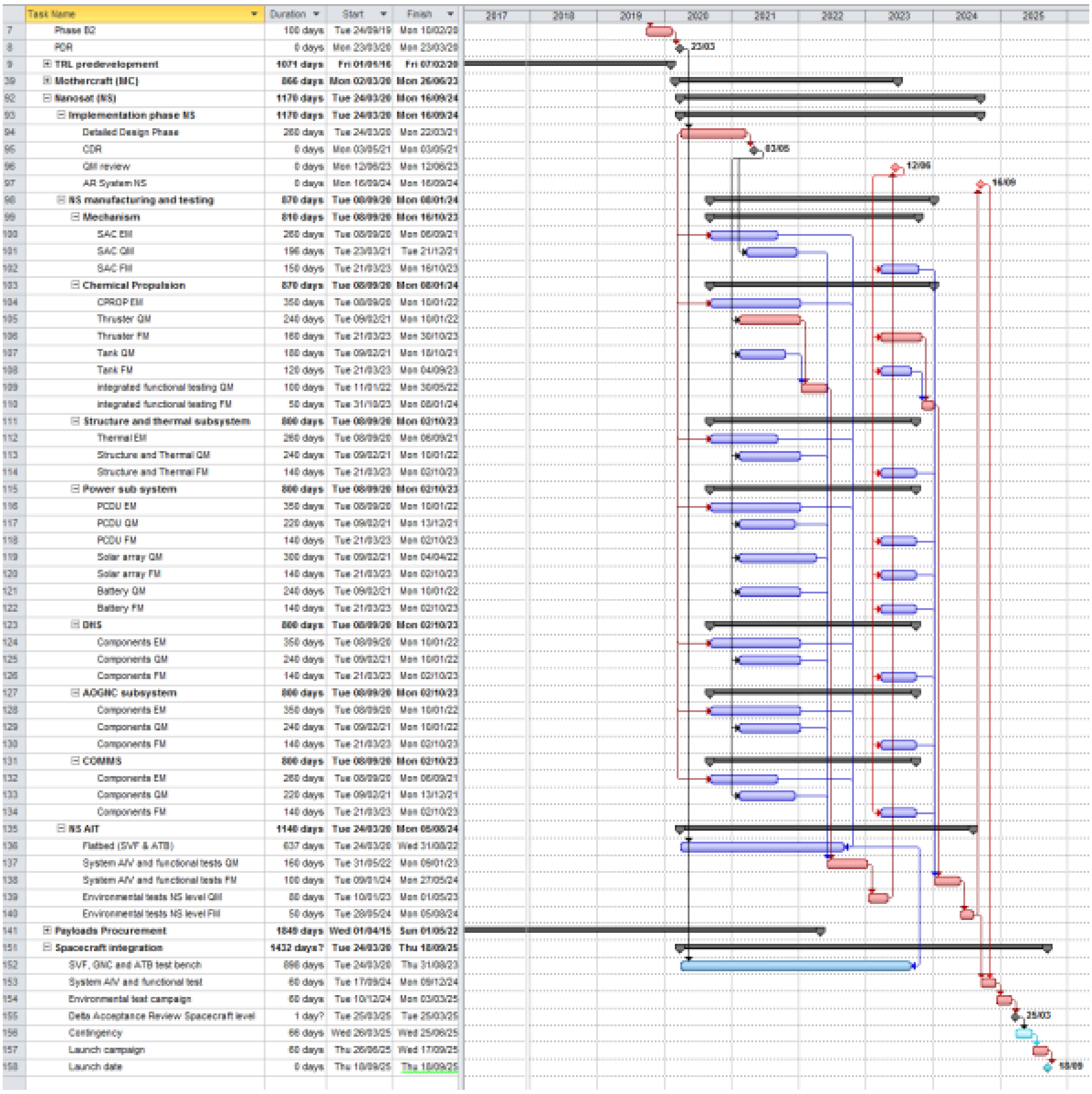

Figure 4-2: Baseline schedule: SS and launch date

4.5.3 Optimised SS Schedule

When analysing the baseline schedule it can be seen that the MC is available about 15 months before the FM of the SS.

The baseline schedule was created using development and testing durations for the SS that try to take into account the smaller dimensions of the subsystems and equipment, which makes handling and testing significantly easier.

This alternative schedule is analysed (Figure 4-3) to understand what the impact on the schedule could be when taking into account benefits of the CubeSat approach during the development and AIV phases. This refers to things like short communication paths, quick decision taking, significant reduction in applicable standards, use of COTS components, etc.

Some assumptions for this schedule version are different from those used for the baseline schedule:

Differences in assumptions for optimised SS schedule	
5 (mod)	Subsystem model durations for baseline (incl. equipment level testing): • SS QM 0.7 years • SS FM 0.5 years
7 (mod)	Subsystem CDRs 2 (SS) to 4 (MC) months before system CDR
12 (mod)	ECSS tailored approach for MC and SS
14 (new)	Shortened durations for QM and FM system level assembly and AIV

This results in the following new milestones for the optimised SS schedule:

- MDR (KO phase A) T0
- PDR T0 +26 months
- CDR in May (SS) / T0 + 42 months
- Delivery of MC PFM in T0 +66 months
- QR for SS in September T0 +56 months
- Delivery of SS FM in T0 + 67 months
- Launch date in T0 +78/79 months

The CubeSat approach naturally increases the risk level in a project, on several levels. It can affect reliability, performance, schedule, etc. Since a Science mission has a different risk profile compared to a typical CubeSat mission, measures need to be taken to reduce these risks to an acceptable level. As a first step, it is here assumed that the QM/FM approach is kept. Furthermore, section 4.3 lists a number of options on how the risks inherent to a CubeSat approach can be reduced.

Also for this schedule approach, the critical path for the SS is driven by the chemical propulsion system. It results in a margin of approximately 3 months for other subsystems for the QM, and 1.5 months for the FM

For the development of the ground station, a time window of 4.3 years is available between PDR and launch date.

In the frame of the CDF Study T0 was assumed as 1 January 2018.

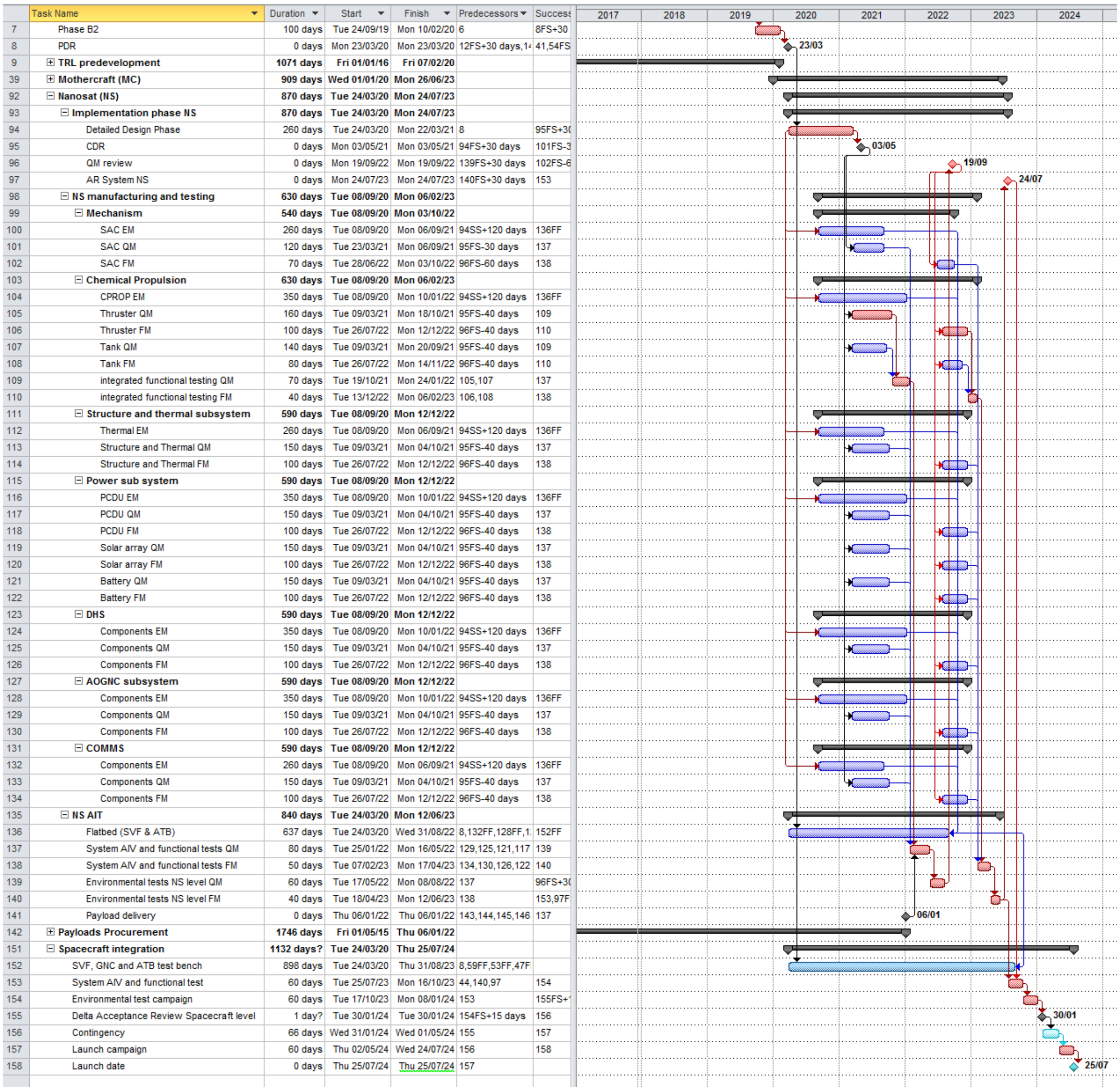

Figure 4-3: SS optimised schedule: SS and launch date

4.5.4 Technology Development Schedule

All technology pre-developments need to lead to a maturity of technology of TRL 6 at the beginning of the implementation phase (PDR). Figure 4-4 shows a graphical representation of all technological pre-development activities as mentioned in section 4.2 and section 4.3.

Light green bars are already ongoing activities, both of them are compatible with an estimated PDR date of beginning 2020.

Orange bars show technology developments that would have had to be started in the past already if they were to be compatible with a PDR date at the beginning of 2020. It is of special importance to highlight these activities because they have a longer duration than the estimated 2 years for Phase A/B1/B2. Independent of the actual date of the

MDR, they would need to be started before the MDR (i.e. before the beginning of Phase A) in order to be compatible with these schedule durations.

Solid green and the shorter blue bars show all the technology developments that are in principle compatible with a schedule duration of 2 years for Phase A/B1/B2. Their technical durations are estimated to be 2 years, thus they should be started latest at the beginning of phase A.

4.5.5 Payload Development Durations

The long blue bars in Figure 4-4 depict, for the baseline schedule, the estimated time available from MDR to the delivery of the (qualification models) of the payloads for integration into the SS. The need date for the payloads is when the qualification model for the spacecraft is being assembled and prepared for system level testing.

In the baseline schedule in Figure 4-4, the payloads have an available development time after MDR of 4 years and 4 months.

The available payload development time in the SS optimised schedule is slightly, though not significantly shorter, namely 4 years exactly (Figure 4-3).

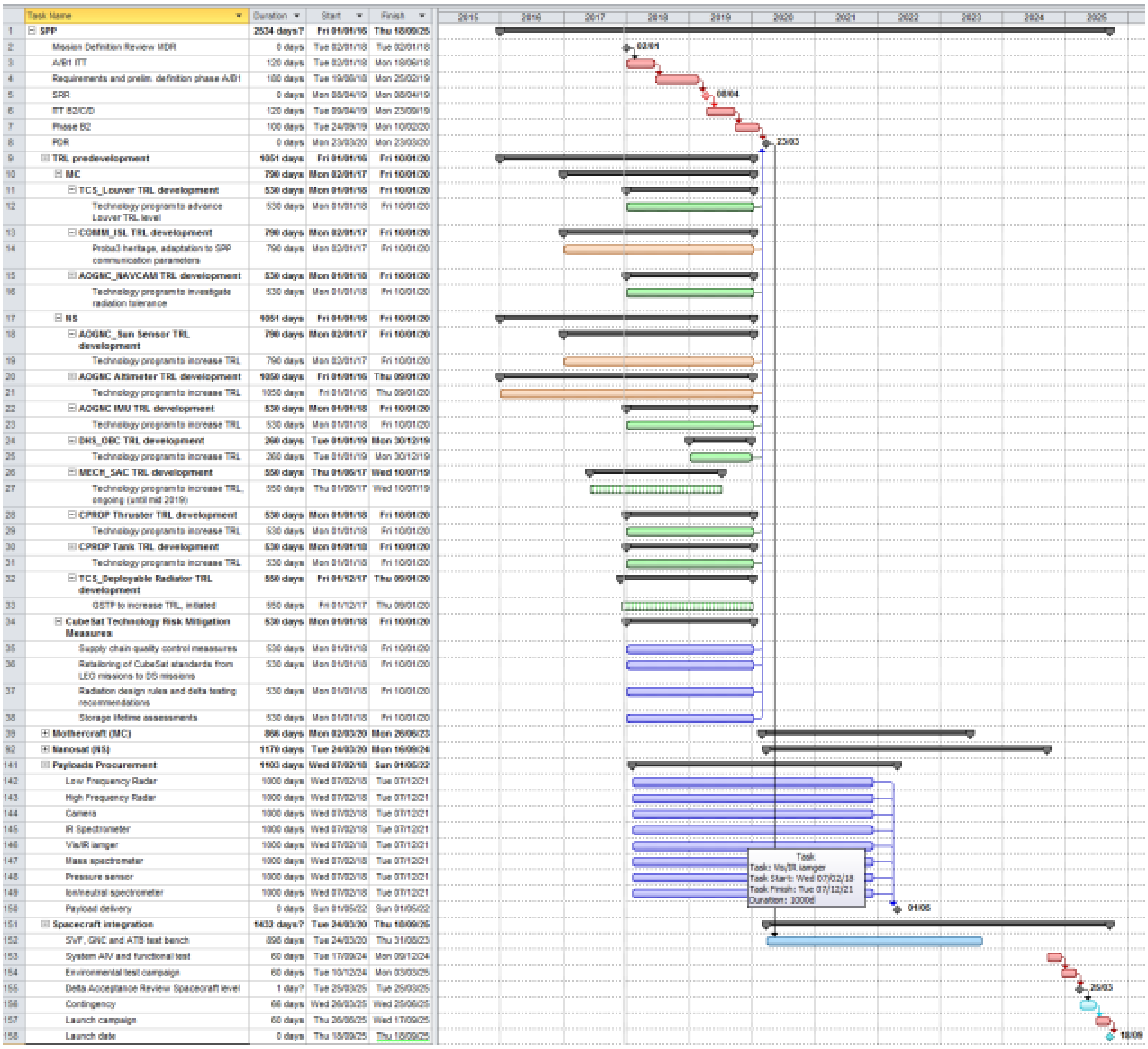

Figure 4-4: Technology developments in baseline schedule and payload delivery dates

4.6 Summary and Recommendations

A project duration (from MDR to launch) of 6.6 years is considered the shortest feasible project duration. This is achieved by adapting, for the SS, a development approach for Cubesats, i.e. smaller and faster with more flexibility with relation to established standards, in conjunction with the use of existing CubeSat technologies. It could thereby be possible to reach a development and AIT duration of 2.5 years for a QM/FM approach for the SS. The MC will use a standard ECSS PFM approach.

If a more conservative approach is taken for the development of the SS, a project duration of 7.8 years from MDR to launch is expected.

The requirement for a launch date between 2024 and 2034 [MIS-070] is confirmed. However, the earliest possible launch date will not be before end of 2024.

3 years of development time for the ground station by ESOC is easily integrated in the schedule for SPP.

Besides using a QM/FM approach for the SS, dedicated activities (see section 4.3) are recommended to help reduce the inherent and increased risks when using CubeSats or CubeSat related technology in order to achieve an acceptable balance for Science deep space missions.

The MC has a high level of maturity making a PFM approach feasible.

The time between kick-off of phase A for the mission and the expected delivery date of the QM of the payload(s) (for integration on a SS) is minimum 4 years.

Starting from the kick-off of phase A, about 2 years are available for pre-development of technologies with TRLs lower than 6 until the PDR. For any technology requiring longer pre-development times than 2 years, the technology pre-development should be initiated correspondingly earlier.

Mission concepts different from SPP (e.g. landers) may require different (additional) environmental verifications compared to “regular” spacecraft. This is not captured in the scheduling and model philosophy.

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5 COST

This chapter presents the cost estimate of the SPP program and it describes the hypotheses and the methodology used.

5.1 Class of Estimate

The cost estimates have been performed within the CDF environment by ESA/ESTEC Cost Engineering (TEC-SYC). The type of cost estimate prepared is Class 4 (as described in the ESA Cost Engineering Chart of Services).

The accuracy of the complete estimate is expected to be +/-20%.

5.2 Cost Estimate Methodology

The cost estimate has been performed in **bottom-up** approach:

- Project office costs of Management, PA, Engineering and AIT (including facilities) assessing the team size at System level and estimating the cost on the basis of tasks durations per phase and European average manpower rates
- HW estimated at equipment level on the basis of:
  - Selling prices or requested price quotations for exiting equipment accounting for Non-Recurring activities in line with 5.4
  - Bottom-up assessment based/benchmarked on experts opinion for units new/delta development and qualifications
  - Analogy to similar equipment/Subsystems/project
  - ESA TEC-SYC cost model suite
  - Expert judgement from CDF technical specialists
- The OPERA TEC-SYC (Latina Hypercube based) cost risk estimation tool.

5.3 Scope of Estimate

The cost estimate includes:

- Industrial cost for implementation phase (B2,C/D&E1) for Mother Craft and Smallsats, including:
  - System level tasks (PO, AIT,V and GSE)
  - Subsystem and units (including delta developments and qualifications from TRL 6 and as identified during the study)
  - Cost risk contingencies shared to Industry
- Launch Services Cost
- Mission Operation Centre costs (MOC), including development
- Science Operation Centre cost (SOC), including development
- ESA internal costs and ESA level contingency.

The cost estimates excludes:

- Small Satellites Instrument suite (assumed as CFI)

- Technology developments identified within the study and addressed in 5.4.

5.4 Main Assumptions

The various requirements and assumptions described in the basic study documentation apply to the cost estimates. In addition, here below are reported the specific cost-related assumptions considered, in particular, for SPP study.

5.4.1 Development Approach

As for the Mother craft, System AIV,T approach based on an EM (ATB) and a PFM; With a Model philosophy at unit level requires EMs and PFMs for avionics.

As for the Smallsat, System AIV,T approach based on an EM (ATB), QM and FM; With a Model philosophy at unit level requires Ems, QM and FMs. Due to very similar design of the Smallsats, a PFM approach has been assumed for the MAIT of 3 Smallsat after the first FM.

Moreover, it has been assumed, that the reliability uncertainty will be successfully addressed through technology and design development activities:

- Procurement agreements for EEE lot and LAT to ensure repeatability
- Successful characterisation and qualification of electronic board to destructive latch-up
- Operational availability (mainly in relation to SEE) to be addressed and mitigated at system level (e.g FDIR) through detailed design development activities
- Low cost approach and related ESA ECSS tailoring to allow CubeSat standard applicability.

5.4.2 Programmatic

In line with the programmatic outcome of the study, the following major assumptions have been accounted:

- Schedule :
  - Phase B2 of 14 months
  - Phase C/D of 40 months
- Launch service: Ariane 6.2 shared Launch (e.g. ARIEL possible primary payload).

5.4.3 Industrial Set-Up

As for the Mother Craft, a lean 3 Tier approach, with 3 S/S outsourced and to be selected during phase B2, has been assumed.

As for the Small Satellites, it has been assumed that a single Prime Contractor will be responsible for the development and AIT,V of the Smallsats, while the subsystem/equipment are assumed “make or buy” (Prime or suppliers responsibility).

5.5 Technology Readiness Level Definition

The Technology readiness levels (TRL) present a systematic measure, supporting the assessments of the availability and maturity of a technology of interest and enabling a consistent comparison in terms of development status between different technologies.

The different levels used in ESA, defined in an internal working group based on NASA's Technology Readiness Levels and ECSS-E-HB-11A are given in the table below:

Technology Readiness Level	Milestone achieved for the element	Work achievement (documented)
TRL 1 – Basic principles observed and reported	Potential applications are identified following basic observations but element concept not yet formulated.	Expression of the basic principles intended for use. Identification of potential applications.
TRL 2 – Technology concept and/or application formulated	Formulation of potential applications and preliminary element concept. No proof of concept yet.	Formulation of potential applications. Preliminary conceptual design of the element, providing understanding of how the basic principles would be used.
TRL 3 – Analytical and experimental critical function and/or characteristic proof-of-concept	Element concept is elaborated and expected performance is demonstrated through analytical models supported by experimental data/characteristics.	Preliminary performance requirements (can target several missions) including definition of functional performance requirements. Conceptual design of the element. Experimental data inputs, laboratory-based experiment definition and results. Element analytical models for the proof-of-concept.
TRL 4 – Component and/or breadboard functional verification in laboratory environment	Element functional performance is demonstrated by breadboard testing in laboratory environment.	Preliminary performance requirements (can target several missions) with definition of functional performance requirements. Conceptual design of the element. Functional performance test plan. Breadboard definition for the functional performance verification. Breadboard test reports.
TRL 5 – Component and/or breadboard critical function verification in a relevant environment	Critical functions of the element are identified and the associated relevant environment is defined. Breadboards not full-scale are built for verifying the performance through testing in the relevant environment, subject to scaling effects.	Preliminary definition of performance requirements and of the relevant environment. Identification and analysis of the element critical functions. Preliminary design of the element, supported by appropriate models for the critical functions verification. Critical function test plan. Analysis of scaling effects. Breadboard definition for the critical function verification. Breadboard test reports.
TRL 6: Model demonstrating the critical functions of the element in a relevant environment	Critical functions of the element are verified, performance is demonstrated in the relevant environment and representative model(s) in form, fit and function.	Definition of performance requirements and of the relevant environment. Identification and analysis of the element critical functions. Design of the element, supported by appropriate models for the critical functions verification. Critical function test plan. Model definition for the critical function

Technology Readiness Level	Milestone achieved for the element	Work achievement (documented)
		verifications. Model test reports.
TRL 7: Model demonstrating the element performance for the operational environment	Performance is demonstrated for the operational environment, on the ground or if necessary in space. A representative model, fully reflecting all aspects of the flight model design, is build and tested with adequate margins for demonstrating the performance in the operational environment.	Definition of performance requirements, including definition of the operational environment. Model definition and realisation. Model test plan. Model test results.
TRL 8: Actual system completed and accepted for flight ("flight qualified")	Flight model is qualified and integrated in the final system ready for flight.	Flight model is built and integrated into the final system. Flight acceptance of the final system.
TRL 9: Actual system "flight proven" through successful mission operations	Technology is mature. The element is successfully in service for the assigned mission in the actual operational environment.	Commissioning in early operation phase. In-orbit operation report.

Table 5-1: Definition of Technology Readiness Levels

5.6 Cost Risk/Opportunity

5.6.1 Definition and Background

In order to define the required cost risk margins at the levels of the Industry costs as well as the ESA internal costs, an ESA internal Monte Carlo-based cost risk assessment tool was applied RD[4]. This standard tool employs triangular cost distributions (Minimum, Most Likely, Maximum) as a simplified but adequate representation of the typically Gaussian cost distributions, requiring a minimum of assumptions as input. The basic cost estimate results at each level (equipment, subsystem, system level activities etc.) are taken as the Most Likely number, i.e. the value with the highest likeliness of occurrence and therefore the top of the cost distribution triangle. The spread from the theoretical absolute Minimum and absolute Maximum cost (both with a probability of occurrence of zero) takes into account various risks and uncertainties, such as the quality and applicability of the references and cost estimate relationships used, the quality of the cost model input parameters, the possible variations in the amount of equipment modifications and qualifications required, market monopoly situations etc. The resulting Cost Risk Margin consists of several components:

- Design Maturity Margin (DMM), to account for additional costs caused by unseen complexities that will be revealed as the design gets into more details. At equipment level these are directly related to the TRL. It is allocated 100% to Industry.
- Cost Model Accuracy (CMA), to account for uncertainties in the cost estimates. It includes the contribution of the Inherent Quality of the cost Models (IQM) together with contextual factors such as the Degree of Adequacy (DOA) of the

cost models used with respect to the specific context of the cost estimate, and the Quality of the Input Values (QIV). Assuming that industry has better and more detailed cost models than ESA because based on internal costs, typically 25% of the CMA is accounted for industry and 75% for ESA.

- Project Owned Events (POE), to account for cost risks induced by potential negative events, as well as potential cost reduction opportunities, that may occur and that are under the direct responsibility of the Project Manager. POE risks are subject to mitigation measures to be managed at Project level. As default, it has been assumed that the POE will be shared 25% for industry and 75% for ESA.
- External to Project Events (EPE), to account for cost risks or opportunities that originate from external influences out of the direct control and responsibility of the Project Manager. The EPE normally belongs 100% to ESA, but ESA regularly transfers the coverage for fair Geo-Return cost impact to Industry. A specific EPE has been included for a higher launch price, as the development of Ariane 6 has only just started, but this ESA EPE is not considered part of the Project budget estimate.

All cost items in the estimate are correlated amongst each other (i.e. the higher cost of one item increases the chance of a cost increase in the other items as well). The resulting Cost Risk Margin has been established for a 70% confidence level: the chance that the budget including the Cost Risk Margin is sufficient for the project is 70%, or in other words the chance of a cost overrun is 30%.

5.6.2 Cost Risk/Opportunity Specific Assumptions

The cost risk parameters used for this study derive from the following considerations:

- Nominal statistical risk assumptions have been made for the MC in relation to the preliminary design status. No particular risks have been identified since the MC is basically a new architecture made of possibly existing[1] units
- For the SS, the technical risks and cost uncertainties, related to the wide usage of COTS components, is mitigated assuming a successful characterisation which is currently planned before implementation phase KO
- 3 months schedule margin is included within the ESA level cost-risk margin for MC and SS.

5.7 Cost Estimate

The cost figures are presented in mid-2017 economic conditions (Note the table is not included in this version of the report).

5.7.1 Mother Craft Industrial Cost

Project Office activities have been estimated on team size assumptions at System and Subsystem level.

[1] Geo-return constraints may led to selection of units characterised by a lower TRL than what has been assumed within the study; however, a risk contingency to take into account these minor impacts (worst case TRL = 6) has been accounted.

AIT/V activities have been estimated on the basis of team size and facilities cost assumptions.

The estimate of the SPP MC Platform has been performed at equipment level, processed with TEC-SYC in house developed and calibrated equipment cost models.

GSE are estimated by parametric `cost to cost` models and "analogy" approach.

The cost-risk margin allocations are summarised in 5.6.2.

5.7.2 Smallsatellites Industrial Cost

The PO and AIT,V cost have been assessed on the basis of the estimated manpower required for the project duration.

The Subsystem costs include the radiation characterisations, new design or delta developments and qualifications as needed.

As for the development and manufacturing of the Smallsats following the first FM, a direct PFM approach, characterised by a much shorter schedule and no need of NREC activities at units level, has been made.

In line with the current preliminary design status the same cost estimate for each of the 3 PFMs has been retained.

5.7.3 SPP Estimated CaC

Launch services costs are based on a shared A6.2 launch.

Mission and Science Centre and Operations Costs are estimated on the basis of the provisional inputs provided by ESOC and ESAC respectively.

ESA Internal cost assumptions have been based on the expected values (average) of similar ESA Projects.

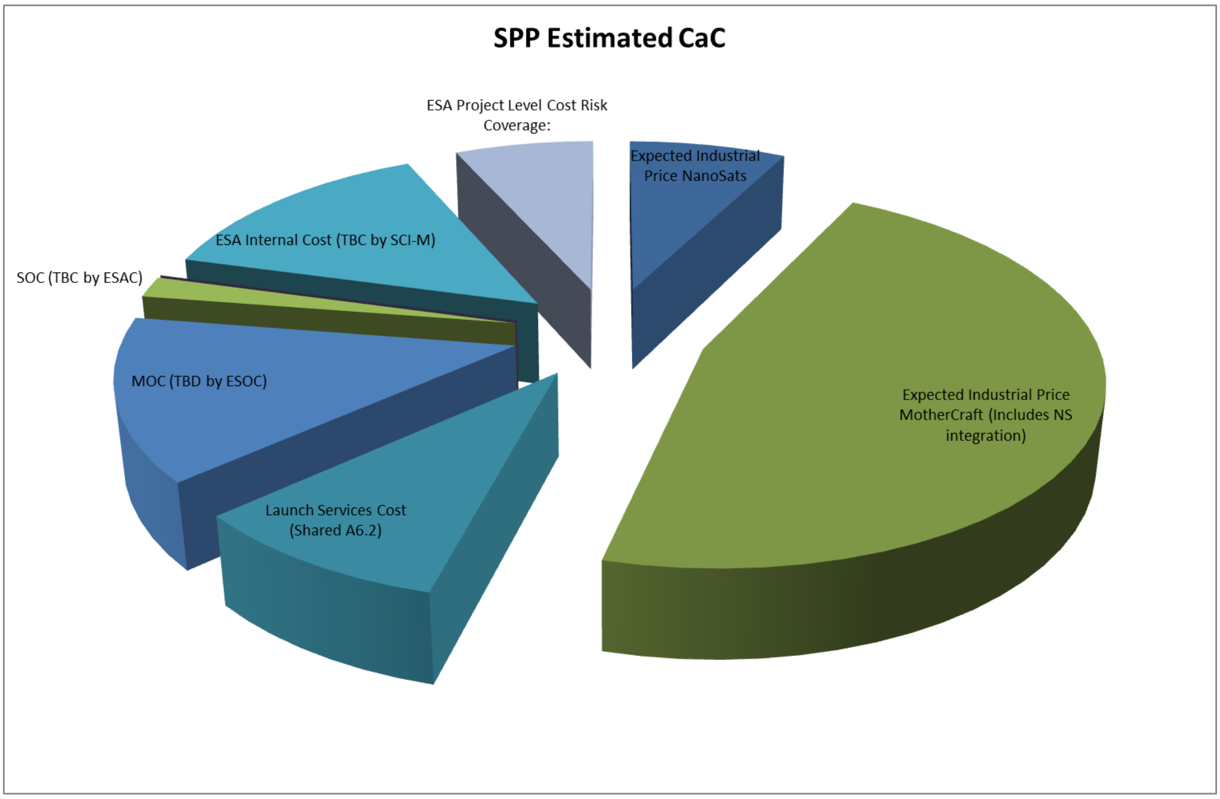

Figure 5-1: Estimated CaC breakdown

5.7.4 NEO Mission Scenario (Option 1) Cost Estimate

For the NEO mission scenario, a major overall cost reduction for the MC is expected due to the lower delta V and power required reflected in lower cost for Solar Array, Electric propulsion and thermal control. While at system level, in line with the overall similarities of the two architectures, similar cost are envisaged.

As for the smallsats, the costs are expected to be higher for the Option 1, mainly due to the deployable radiator included within the smallsats design to achieve the thermal control required. The bigger solar array will also have an impact. It is highlighted that these aspects will most probably have an impact at system level due to the overall more complex system design.

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6 CONCLUSIONS

6.1 Achievement of Study Objectives

CDF Study Objectives have been addressed and achieved, as described hereafter:

#	STUDY OBJECTIVE	ACHIEVED
1	Assess the feasibility of performing deep space planetary missions with an architecture consisting of a mothership spacecraft carrying a swarm of smallsats to be deployed for multi-point science observations. Main goal is not to design a specific mission but to provide a “tool-box” of technical building blocks that the community can use to develop new planetary missions architectures, in reply to future science calls.	YES
	How it was addressed: Study cases selection enabled tool-box definition. Several sensitivity analyses performed.	
2	Highlight the main operational constraints (i.e. max communication range vs achievable data rates, communication links between the mothership and the swarm, max number of smallsats, etc) imposed by the architecture, identifying technical solutions for a variety of scenarios including: a) Rendez-vous missions to small bodies, b) Missions around Mars (Phobos/Deimos) and Venus.	YES
	How it was addressed: By design / sensitivity analysis / trade-offs / references to similar missions (whenever possible).	
3	Investigate the required adaptation of existing commercial platforms for use in deep space missions and identify any new specific technology developments enabling missions.	YES
	How it was addressed: SPP Team includes consultants with extensive experience in smallsats. Structured set of information produced, to be used as input for Technology Roadmap formulation (to be refined, taking into account European capabilities and geo-return constraints).	
4	Carry out preliminarily design of the mothercraft and the smallsats	YES

#	STUDY OBJECTIVE	ACHIEVED
	and perform parametric analyses to understand the flexibility/ adaptability of the design to various environments.	
	How it was addressed: By design / sensitivity analysis / trade-offs / formulation of study cases and sequence in which they have been investigated.	
5	Assess the possibility of adding a lander asset on the surface of the small body.	YES
	How it was addressed: Study Session dedicated to high level sensitivity assessment for additional scenarios (Phobos/Lander), including drivers, criticalities, scalability considerations. Delta Session dedicated to Multi-targets mission concept).	
6	Provide a portfolio of potential transfers to small bodies for launches between 2024 and 2034.	YES
	How it was addressed: Mission analysis trade-offs plus collection and plans for realisation of a web based repository of all relevant transfers studied in previous exercises, i.e.: M-ARGO, AIM, Marco-Polo etc. for science and industrial systems teams to have background info (*date dependency will be highlighted).	
7	Define the programmatic approach, including the procurement of the smallsats as part of the payload complement.	YES
	How it was addressed: The SPP Team includes a programmatics expert and consultants with extensive experience in smallsats.	
8	Assess the mission cost, with a target to be below an M-class (ideally around 150M€).	YES
	How it was addressed: SPP Team includes a Cost expert.	
9	Study the implications of this mission architecture for mission operations.	YES
	How it was addressed:	

#	STUDY OBJECTIVE	ACHIEVED
	Information on operation strategy adopted by ESOC, including constraints etc. Heritage from previous missions.	

6.2 Main Findings

The CDF SPP Study has identified feasibility boundaries - at system level - for the mission concept.

For Option 1 (NEO – Inactive Body), 4 smallsats carrying a payload mass of around 3kg can be transferred to the selected target within 5 years by:

- A shared launch with Ariane 6.2 to L2, with electric propulsion (T6 or PPS1350) transfer
- A shared launch with Ariane 6.2 to GTO, escape with chemical propulsion and transfer with electric propulsion (T6 or PPS1350)
- A dedicated launch with EPSILON to 200 x 4500 km orbit, with electric propulsion (T6) transfer (*Note: design assumed as Option 2 one: could be optimised further, thus shortening transfer time which is longer than 4 years in the table and increasing payload mass which is marginal at the moment: 3.11 kg*). Open points for this option are identified at System level in this Executive Summary.

A shared launch with Ariane 6.2 to GTO, followed by escape and transfer based on chemical propulsion, could deliver around 2.42 kg of payload at the target within 2 years. This option could become appealing should a smaller payload mass represent an attractive science case or if the number of smallsats would be reduced to 3 or less.

Target	CP Engine	EP Engine	Launcher	Mass at Target	PL/MC	#NS	Pl/NS	Payload Mass for 1 NS	Time of Flight (Years)
NEOs	CP	-	Epsilon	92.79	20%	4	12%	0.56	2.00
NEOs	CP	-	VegaC	132.62	20%	4	12%	0.80	2.00
NEOs	CP	-	Ariane 6.2 GTO	402.61	20%	4	12%	2.42	2.00
NEOs	CP	-	Ariane 6.2 L2	43.10	20%	4	12%	0.26	2.00
NEOs	-	T6	Epsilon	864.89	12%	4	12%	3.11	4.20
NEOs	-	PPS1350	Epsilon	539.89	12%	4	12%	1.94	4.76
NEOs	-	T6	VegaC	1539.84	12%	4	12%	5.54	8.16
NEOs	-	PPS1350	VegaC	921.52	12%	4	12%	3.32	9.28
NEOs	-	T6	Ariane 6.2 GTO	1514.93	12%	4	12%	5.45	6.26
NEOs	-	PPS1350	Ariane 6.2 GTO	1015.76	12%	4	12%	3.66	7.00
NEOs	-	T6	Ariane 6.2 L2	752.95	20%	4	12%	4.52	2.10
NEOs	-	PPS1350	Ariane 6.2 L2	582.48	20%	4	12%	3.49	2.14
NEOs	CP	T6	Epsilon	165.12	20%	4	12%	0.99	2.07
NEOs	CP	PPS1350	Epsilon	153.44	20%	4	12%	0.92	2.12
NEOs	CP	T6	VegaC	235.99	20%	4	12%	1.42	2.11
NEOs	CP	PPS1350	VegaC	219.30	20%	4	12%	1.32	2.18
NEOs	CP	T6	Ariane 6.2 GTO	716.44	20%	4	12%	4.30	2.32
NEOs	CP	PPS1350	Ariane 6.2 GTO	665.78	20%	4	12%	3.99	2.54
NEOs	CP	T6	Ariane 6.2 L2	76.69	20%	4	12%	0.46	2.03
NEOs	CP	PPS1350	Ariane 6.2 L2	71.27	20%	4	12%	0.43	2.06

Table 6-1: Option 1 – NEO - Inactive Body

For Option 2 (Main Asteroid Belt – Active Body), 4 smallsats carrying a payload mass of around 3kg can be transferred to the selected target within 5 years by:

- A shared launch with Ariane 6.2 to L2, with electric propulsion (T6) transfer
- A shared launch with Ariane 6.2 to GTO, escape with chemical propulsion and transfer with electric propulsion (T6 or PPS1350).

The target distance (2.5 A.U.) reduces the options available to implement such a challenging scenario.

Target	CP Engine	EP Engine	Launcher	Mass at Target	PL/MC	#NS	PI/NS	Payload Mass for 1 NS	Time of Flight (Years)
Main Asteroid Belt Inner	CP	-	Epsilon	37.69	20%	4	15%	0.28	2.00
Main Asteroid Belt Inner	CP	-	VegaC	49.03	20%	4	15%	0.37	2.00
Main Asteroid Belt Inner	CP	-	Ariane 6.2 GTO	157.40	20%	4	15%	1.18	2.00
Main Asteroid Belt Inner	CP	-	Ariane 6.2 L2	12.04	20%	4	15%	0.09	2.00
Main Asteroid Belt Inner	-	T6	Epsilon	761.53	12%	4	15%	3.43	6.58
Main Asteroid Belt Inner	-	PPS1350	Epsilon	449.61	12%	4	15%	2.02	6.68
Main Asteroid Belt Inner	-	T6	VegaC	1348.49	12%	4	15%	6.07	12.36
Main Asteroid Belt Inner	-	PPS1350	VegaC	764.75	12%	4	15%	3.44	12.56
Main Asteroid Belt Inner	-	T6	Ariane 6.2 GTO	1340.35	12%	4	15%	6.03	10.51
Main Asteroid Belt Inner	-	PPS1350	Ariane 6.2 GTO	842.97	12%	4	15%	3.79	10.62
Main Asteroid Belt Inner	-	T6	Ariane 6.2 L2	674.84	12%	4	15%	3.04	4.28
Main Asteroid Belt Inner	-	PPS1350	Ariane 6.2 L2	483.39	12%	4	15%	2.18	4.22
Main Asteroid Belt Inner	CP	T6	Epsilon	135.81	20%	4	15%	1.02	2.14
Main Asteroid Belt Inner	CP	PPS1350	Epsilon	116.87	20%	4	15%	0.88	2.23
Main Asteroid Belt Inner	CP	T6	VegaC	176.69	20%	4	15%	1.33	2.19
Main Asteroid Belt Inner	CP	PPS1350	VegaC	152.06	20%	4	15%	1.14	2.30
Main Asteroid Belt Inner	CP	T6	Ariane 6.2 GTO	567.16	20%	4	15%	4.25	2.60
Main Asteroid Belt Inner	CP	PPS1350	Ariane 6.2 GTO	488.10	20%	4	15%	3.66	2.96
Main Asteroid Belt Inner	CP	T6	Ariane 6.2 L2	43.39	20%	4	15%	0.33	2.05
Main Asteroid Belt Inner	CP	PPS1350	Ariane 6.2 L2	37.34	20%	4	15%	0.28	2.07

Table 6-2: Option 2 – Main Asteroid Belt – Active Body

Major design constraints have been highlighted – at systems and subsystems level, as described in detail in the technical chapters of the SPP Report.

The following points deserve particular attention:

- The ”standard” Margin Policy typically used for classical science satellites may not always be applicable to smallsats and needs to be revisited (example: AOGNC Delta V margins and Systems margins are not adequate to the smallsat platform “size”, producing an overdesign which would be unnecessary)
- Scalability considerations (both the scaling-down from bigger platforms and the scaling-up from Cubesats) are not always directly applicable to the smallsats design as well as the relevant deploying mechanisms. Careful analysis is recommended on a case-by-case basis
- Volume and shape factor are the biggest drivers for the smallsat design, rather than mass, in particular to avoid the design of dedicated equipment. Consequently:
    - Power is limited by the size and shape factors of the platform, and this imposes constraints, particularly on the instrument as well as in the ISL communication system.
- Thermal dissipation is critical because the radiator area is limited by the platform reduced size. A careful optimization of the payload duty cycles would help to lower the thermal dissipation requirements. Additionally, a detailed trade-off between adding deployable radiators (for the NEO option) or having larger surfaces (i.e. a bigger smallsat) should be considered in the future. Adopting

existing design solutions, which allow respecting the standard “form factors” saves costs, however detailed Trade-Offs shall confirm the exclusion of dedicated design.

- ISL between the mother spacecraft and the smallsats does not represent a limitation with current assumed distances (MC-*SS relative geometry: MS-Target 12-20km, SS-Target 5-16km*). Detailed CONOPS, based on the mission profiles, would help refining the link budgets, which is expected to be not lower then 30kbps as average exchange rate for a single smallsat).
- For the MC-SS ISL link, a “Star” topology (MS as centre) has been selected as baseline. If the geometry boundary conditions change (in particular if the target size increases) the ISL “Mesh” architecture could be an enabler for the inter-satellite-link capability.
- Target size and gravity knowledge drive the minimum distance achievable for the SS from the target, with implications on Flight Dynamics, GNC and Operations (for example: *if the target size is > 1km and the required altitude from target is ~5km, an orbit would be needed instead of hyperbolic arcs – to be confirmed by detailed analysis including cost, science objectives, delta Vs, communications with MC*). A bigger target would imply a more complex system, and more expensive operations).
- Whenever it is possible to operate the SS at target adopting hyperbola arcs, a higher flexibility is obtained compared to standard orbits:
  - The hyperbolas can be placed to cover specific target sites to be observed, without the expensive orbital inclination changes
  - The hyperbolas can be conceived to optimise Sun aspect angle, helping Solar Arrays and radiator accommodation
  - The hyperbolas offer optimal visibility conditions for the inter-satellite-link MC-SS, as explained in detail in the AOGNC chapter.

Last, it has to be highlighted that:

- The operational complexity for the mission concept is rather high as the architecture includes 5 spacecraft
- The design of the MC is challenging, as the platform has to cope with very different environments (for example: Thermal Design for MC is very complex as it has to withstand high dissipation at Earth and high heating power at target)
- A common design for the SS would reduce development time and cost, however scientific objectives require different payloads to be embarked on the smallsats which may imply design variations at platform level.
- The TRL for the smallsat developments is rather low (small platform are developed for LEO but not for interplanetary applications); starting the SS developments before MC would mitigate risks, however MC interfaces would have to be considered for the SS design. A development strategy shall be duly assessed and adopted.

- Finally it shall be taken into account that small bodies may have significant orbit changes due to the influence of larger bodies. This shall be considered when selecting the target and the launch date.

6.3 Further Study Areas

The SPP Study has identified areas which are considered of high interest, however due to time limitations a detailed assessment could not be performed. It is recommended to further investigate the following:

- The trajectory optimisation process should continue in the future in order to assess the benefits of adding Earth and/or Mars gravity assists manoeuvres. These would reduce the required total DV (by increasing the operational time and complexity) and could also widen the number of reachable targets (higher inclinations in the main belt could potentially be targeted).
- Propulsion strategy for the transfer to a NEO target, including a Kick-Stage and electric propulsion. This Option was not retained as baseline, based on observations derived from previous CDF studies. However a detailed assessment of pros and cons of adopting a propulsion module would be beneficial to the study in particular for targets with a longer transfer time.
- Thermal Design for the MC, accounting for the aperture of the “doors” releasing the smallsats. Current design is simplified and does not consider the impact of this event in the mission timelineAdding payload on the mothercraft, which could host instruments taking measurements not requiring multiple-point observations. As a working assumption, in the CDF SPP Study no payload was considered on the MC. However there would be a lot of power available, once at target, as the electric engine would no longer be operated. This power could be used for scientific instruments. This consideration must be taken very carefully since adding payload on the mothercraft would certainly make its design more complex and heavier and would therefore detrimentally impact the mass resources available for the smallsats. The best distribution of payload must be proposed by the scientific community keeping in mind that the purpose of a mission with smallasat would be to have very focused science objectives benefiting from the capability of the swarm of smallsats to perform multi-point observations while operating close to the target body surface.
- Jets and outgassing impact on SS observing active bodies (STR blinding etc) shall be assessed, along with design strategies potentially required to mitigate the detrimental effects.
- SS Thermal design optimisation including synergies with AOGNC strategy. Dedicated pointing manoeuvres could simplify the design
- MC and SS power design optimisation by looking at, for example, alternative solar array technologies such as flexible arrays, for example.
- Interface with the EPSILON Launcher to be refined, including a dedicated design for this option. A system level assessment has been performed taking into account the design performed for CDF SPP Option 2, and there are open points to be addressed (*Example: with 2m HGA, EPSILON would not offer required volume*)

Radiation effects (*doses and proton induced single event effects*) in the case of launch with EPSILON or VEGA in LEO, followed by an electric propulsion transfer. Design measures, shielding requirements and selection of specific equipment shall be identified as well as launch orbit optimization for reducing the spiral-out phase.

- Accommodation details in the smallsats, depending on the specific instrument selection and on the consequent specific platform needs (*example: deployable radiators for Option 1*)
- "Cubesat" equipment procurement methods, including strategies ensuring adequate PA. Origin and quality control are issues at the moment and shall be properly addressed in order to define a suitable approach for the considered type of mission.
- Reliability strategy to be adopted for MC and SS, based on technology readiness and risks levels considered acceptable at mission level shall be reassessed.

The boundary conditions identified by the CDF SPP activity are valid under the assumptions taken in the course of the study and aim at identifying inter-dependencies, order of magnitudes, ball-park numbers and areas for further assessment and development.

Detailed analysis is instrumental to confirm results, in particular based on:

- Specific target selection
- Scientific Payload definition
- Risks, Programmatics, Cost considerations (including potential co-operations).

6.4 Final Considerations

The CDF Study did not focus on optimising the design for a specific mission, but provided a structured collection of technical solutions, constraints and building blocks to develop planetary mission architectures.

Two reference study cases (Option 1 & Option 2) were selected, offering the boundaries of a vast trade-space explored.

Sensitivity analysis and trade-offs (at system and subsystem level) within the 2 study cases provided order of magnitudes for the sizing parameters and identified the design drivers.

Synthesis of the results at system level and collection of transfer's data from previous missions will be available to science and industrial systems teams to have background info, based on already relevant preformed assessments (a *web based repository of all relevant transfers studied in previous exercises will be created*).

The study offered indications for Technology Requirements, useful as inputs for Technology Roadmaps formulation, after refinement including European capabilities and geo-return constraints.

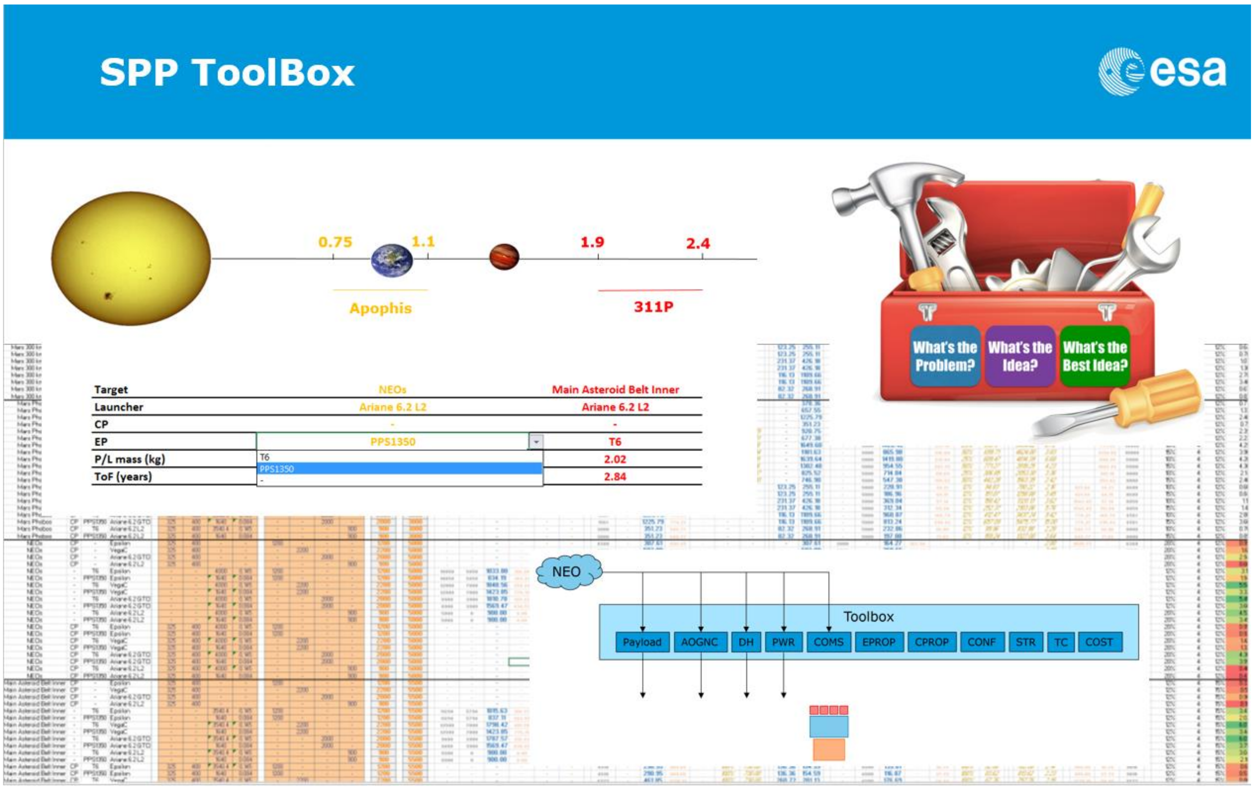

Figure 6-1: SPP Toolbox

Ultimately, SPP produced a toolbox useful to develop new planetary missions architectures, in reply to future science calls.

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7 REFERENCES

7.1.1.1 Chapter 3 Mission Options

RD[1] *Minor Planet Center* database, http://www.minorplanetcenter.net/data

7.1.1.2 Chapter 4 Programmatics References

RD[2] Technology Readiness Levels Handbook for Space Applications, ECSS-E-HB-11A, Issue 1, Dated March 2017

RD[3] Space Engineering, Verification guidelines ,ECSS-E-HB-10-02A, Dated December 2010

7.1.1.3 Chapter 5 Cost References

RD[4] TEC-SYC Cost Risk Procedure, TEC-SYC/5/2010/PRO/HJ, February 2010

RD[5] ESA Cost Engineering Charter of Services, Issue 4, TEC-SYC/12/2009/GRE/HJ

RD[6] "Guidelines for the use of TRLs in ESA programmes", ECSS-E-HB-11A dated 01-03-2017

RD[7] ECSS-E-AS-11C Adoption Notice of ISO 16290, Space Systems – Definition of the Technology Readiness Levels (TRLs) and their Criteria of Assessment, dated 1 October 2014. To be supersede by EN16603-11.

RD[8] CDF Study Reports, CDF-178(A), December 2017 and CDF-178(B), December 2017

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8 ACRONYMS

Acronym	Definition
AIT/V	Assembly, Integration and Test/Verification
AIV	Assembly, Integration and Verification
AOCS	Attitude, Orbit Control System
AST	Advanced Space Technologies GmbH
ATB	Avionics Test Bench
AU	Astronomical Unit
AVM	Avionics Model
AVUM	Attitude Vernier Upper Module (VEGA Upper Stage)
CaC	Cost at Completion
CAM	Collision Avoidance Manoeuvre
CDF	Concurrent Design Facility
CDR	Critical Design Review
CER	Cost Estimation Relationship
CFI	Customer Furnished Instruments
CMA	Cost Model Accuracy
COTS	Commercial Off The Shelf
CP	Chemical Propulsion
DHS	Data Handling System
DLL	Design Limit Load
DM	Development Model
DML	Declared Materials List
DMM	Design Maturity Margin
DOA	Degree of Adequacy of the cost model
DoF	Degrees of Freedom
DOR	Differential One-way Ranging
DPL	Declared Processes List
ECSS	European Cooperation on Space Standardisation
EDRS	European Data Relay Satellite
EGEP	Enhanced Galileo Electric Propulsion

Acronym	Definition
EM	Engineering Model
EMC	Electro Magnetic Compatability
EO	Earth Observation
EP	Electric Propulsion
EPE	External Project Events
EQM	Engineering and Qualification Model
FCU	Fuel Control Unit
FDIR	Failure Detection, Isolation and Recovery
FM	Flight Model
FoV	Field of View
GEO	Geostationary Equatorial Orbit
GNC	Guidance, Navigation and Control
GSE	Ground Support Equipment
GSP	General Studies Programme
GTO	Geostationary Transfer Orbit
HDRM	Hold Down and Release Mechanism
HGA	High Gain Antenna
HPR	High Pressure Regulator
HW	HardWare
IMU	Inertial Measurement Unit
IQM	Inherent Quality of the cost Model
IR	Infra Red
ISL	Inter Satellite Link
ISO	International Organisation for Standards
ISS	International Space Station
ITAR	International Traffic in Arms Regulations
ITT	Invitation to Tender
LEO	Low Earth Orbit
LoS	Line of Sight
LV	Launch Vehicle
MAB	Main Asteroid Belt

Acronym	Definition
MAIT	Manufacturing Assembling Integrating Testing
MC	MotherCraft
MDR	Mission Definition Review
MGA	Medium Gain Antenna
MLI	Multi-Layered Insulation
MM	Memory Module
NEA	Near Earth Asteroid
NEO	Near Earth Object
OBC	On-Board Computer
OCDT	Open Concurrent Design Tool
PCDU	Power Conditioning and Distribution Unit
PDR	Preliminary Design Review
PFM	Proto-Flight Model
PI	Principal Investigator
POE	Project Owned Events
PSCU	Power Supply and Conditioning Unit
PPU	Plasma Propulsion Unit
QIV	Quality of the Input Values
QM	Qualification Model
RTU	Remote Terminal Unit
RW	Reaction Wheel
SAC	Solar Array Controller
SADM	Solar Array Drive Mechanism
SMA	Shape Memory Alloy
SS	SmallSats
S(T)M	Structural (Thermal) Model
SPP	Small Planetary Platforms
STM	Structural Thermal Model
SVF	Software Validation Facility
SVM	Service Module
TM	Thermal Model

Acronym	Definition
TRL	Technology Readiness Level
TVAC	Thermal Vacuum (Test)

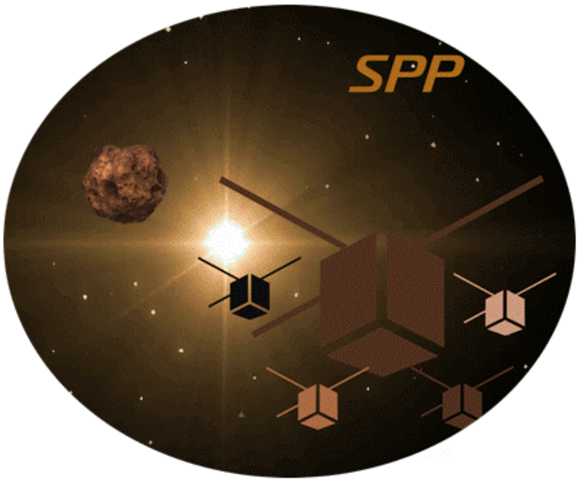
A MULTI-ASTEROID TOURING CONCEPT

SPP

Delta- Session on Multi Asteroid Touring Concept

External Final Presentation
ESTEC, 07-03-2018

Prepared by the CDF* Team



(*) ESTEC Concurrent Design Facility

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Multi-Target Concept

- The idea of a multi-target mission was proposed by P. Jahunen in the frame of the New Science Ideas call.
- This concept has been studied in one dedicated delta session of the SPP CDF study.
- The main science goal is the characterisation of a "statistically significant" number of asteroids through fly-bys and remote sensing observations (visible + IR spectro)
- As there can be huge variability in the definition of a "statistically significance", it was clarified with the scientific community that the range of 10 to 100 asteroids was to be considered for the purpose of this study:
  - 10 being the absolute minimum and 100 being the desired number.

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Multi-target Mission architectures

- Several mission architectures could be envisaged to fulfill the established mission objectives, i.e.: one satellite visiting many targets of several satellites visiting targets, mother+daughters architectures vs only small satellites travelling to the Main Belt, etc
- Because of the limited time for the study, it was decided to focus on reusing the concept studied in the main SPP sessions as much as possible.
- The results of this option were compared with the original proposal featuring multiple nano spacecrafts travelling to the Main Belt with e-Sail propulsion.

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Concept 1 – Pekka Janhunen's Proposal

- A swarm of nanosats flying each one independently to the main belt with E-Sail.
- E-sail propulsion - **50 nanosats of 5 kg** visiting each one **6 asteroids** in average and flying by Earth on the return where they would downlink Science data
- Mission duration: **3.2 yrs**
- CDF Assessed: feasibility of a low-thrust trajectory (with off the shelf electric propulsion), allowing to fly by several asteroids (5-6 in the proposed paper) and then flying by Earth for science data return

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CDF Session 9 Scope

- CDF Session 9 focused on the assessment of Concept 1, indicating weak points and validating feasibility of the proposed technical solution
- A number of disciplines have been asked to assess the mission concept described by Pekka Janhunen:
  - Mission Analysis
  - Operations
  - GNC
  - Electric Propulsion
  - Comms
  - Systems

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Concept 1 - Mission Options

P. Janhunen	CDF Baseline	FULL
Full autonomy	Partial autonomy	Ground Based OPS
NAV + Manoeuvres	NAV	JUICE heritage
No COMMS Data download @ Earth Fly-by	COMMS HK only Data download @ Earth Fly-by	COMMS HK + NAV Data download @ Earth Fly-by

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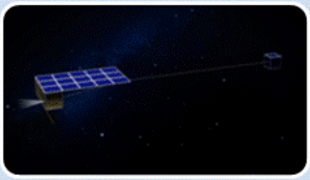
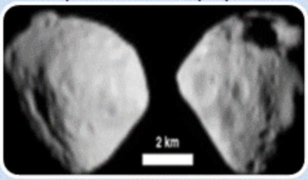
Pekka Janhunen's 2016 Proposal Summary (before IEEE update)

Overview



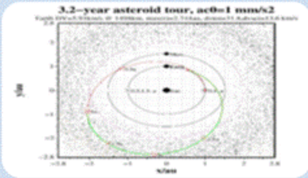
- Baseline: swarm of nanosats flying independently to Main Belt with low thrust propulsion
- Statistical study of geology and geophysics of targets
- 50 nanosats of 5 kg
- Each nanosat visits 6 asteroids in average
- Data stored in Flash memory and downloaded by an Earth fly-by at the end
- Cost of 60 million Euros

Population Geophysics



- Emphasis on small bodies Spectral data from many asteroids
- Sizes, spectral classification, and albedos for small MBOs (1 km < D < 20 km)
- 4-centimeter telescope for surface imaging w/ resolution of 100 meters or better
- At any given time, in the MB there are 5 or more asteroids within 10M km
- Sci data volume: 10GB per s/c
- Max data rate needed 10Mb/S

E-Sail



- Nanosatellites propelled by innovative electric solar wind sails
- Each s/c carries a single Coulomb drag tether
- Flybys of targets in the Main Belt at a range of around 1000 km
- Acc @ 1 au of 1 mm/s2 with a constraint of max of +/- 30° in direction of sun
- Mission duration: 3.2 yrs
- Total DV: 31.8 km/s
- @ 1000 km, 10km/s relative speed FB nearest duration – 2m
- Each year in MB =4,5 flybys/s/c

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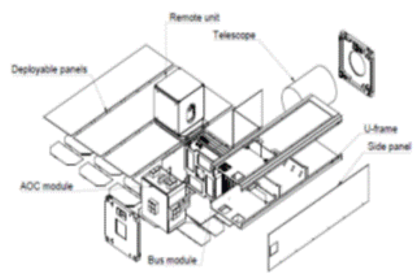
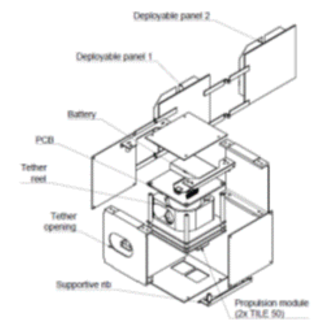
IEEE Paper MTA updates

- Sci data volume: 50GB per s/c
- 20h contact with Ground Station per s/c
- 4cm telescope replaced by 8cm main telescope (for viewing asteroids during FB + assistance for star tracking and optical navigation)+ 2cm framing camera (for optical navigation and tracking celestial objects)





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Systems Presentation

Confirmed/Open points

- Confirmed by MA that any given time, in the MB there are 5 or more asteroids within 10M km
- Need of optical and autonomous navigation for close encounter
- In a first assessment - same assumptions of Pekka's proposal regarding trust to mass ratio and thrust direction (acceleration around 1 mm/s2, constrained thrust direction within 30deg of radial direction) then different levels of thrust to mass ratio and no constraint on the optimal thrust direction were considered
- MTA proposal:
  - COMMS solution feasibility for s/c and RU
  - Performance characterization of telescope + auxiliary camera used for viewing the asteroids during FB and as star tracker
  - Thermal and structural analysis
  - SCI limitation

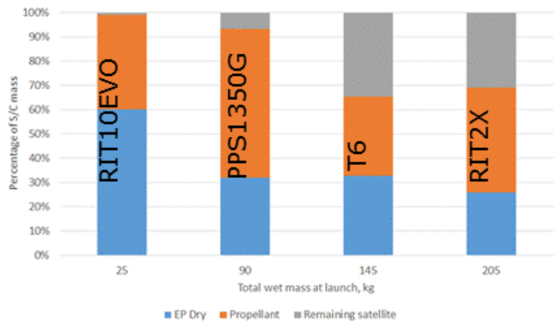
Assumptions 1

- Reproducing the trajectories in the MTA proposal
- High thrust to mass ratio: acceleration @ 1AU 1 mm/s2
- Thrust direction within 30deg of radial direction
- Transfer Δv = 15.5 km/s

- For EP input:
  - Power not a constraint
  - No TVC
  - No CGT
  - No redundancy
  - Ideal thruster operation setpoint
  - EP Dry = Thruster + Tank + PPU + Fluid Management (TRL4-5 for RIT10 case)

SS Inputs 1

- MA: Transfer Δv = 15.5 km/s (Warning - using the Delta-V provided by the E-sail, which has certain constraints, to assess the performance of conventional EP engines. With conventional EP much less Delta-V is needed)
- EP:



- COMMS: M-Argo as reference for HGA
- GNC: JUICE as reference for NAVCAM

Mass Budget 1

For Pekka: 0.5kg MEMS CG + M-Argo + JUICE NAVCAM

Includes 20 km tether (100g) + high voltage source + electron gun (1kg)

DHS/MECH/PWR/THERMAL **ref. M-Argo ?**

5kg S/C ref. from EPCS paper

40% of dry mass

Mass Budget		S/C [kg] Adpt.Pekka1	S/C [kg] Adpt.SPP1
Attitude, Orbit, Guidance, Navigation Control		7.70	14.80
Communications		3.00	3.00
Chemical Propulsion		0.00	0.00
Data-Handling		0.25	3.60
E-sail		1.10	0.00
Electric Propulsion		0.00	88.54
Instruments		0.60	0.60
Mechanisms		0.59	15.07
Power		2.55	136.78
Structures		2.61	52.34
System Engineering		0.00	0.00
Thermal Control		0.78	69.90
Harness	5%	0.96	19.23
Dry Mass w/o System Margin		**20.14**	**403.86**
Dry Mass w/ System Margin		**24.17**	**484.63**
EPROP Propellant Mass		0.00	201.93
EPROP Propellant Residual		0.00	4.04
Total Wet Mass		**24.17**	**690.60**

x29

JUICE NAVCAM | SPP – 2.4kg

M-Argo HGA

22% of dry mass

Inst. Aalto-1 (spectral imager ~ small NIR-optical inst.)

33% of dry mass

15% of total dry mass

A-SPP values ***from SPP OPT2*** *with corrections – no redundancy, no deployers*

50% of total dry mass (from T6 graph.)

MA Updates Synthesis

- Electric sail acceleration of 1 mN/kg in a direction up to 30 deg from the radial -> drove the design - large Delta-V 15.5 km/s (works in proposal case – no propellant)
- Removing constraint - thrust much more efficiently in the tangential direction + less Delta-V 7 (1 mN/kg) - 10.5 km/s (0.1 mN/kg)
- To manoeuvre in the belt and target different asteroids up to 3.15AU - minimum acceleration of 0.025 mN/kg (@ >3 AU)
- New input from EP (less demanding requirements)

Assumptions 2

- Preliminary assessment
- Initial acceleration of 0.2 mN/kg
- Total Δv = 10.5 km/s (including ~9.5 km/s for the transfer (in line with the results obtained for 0.25 mN/kg = 9.1 km/s) and 1 km/s for the fly-bys of the asteroids)

- For EP input:
  - Power not a constraint
  - No TVC
  - No CGT
  - No redundancy
  - Ideal thruster operation setpoint
  - EP Dry = Thruster + Tank + PPU + Fluid Management (TRL4-5 for RIT10 case)

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Mass Budget updates from IEEE paper

- The spacecraft bus and E-sail components (high-voltage source, reel, motor) are scaled from the ESTCube-2 design

- 5x TILE 50 modules in the main s/c to provide three rotations (two directions each) and one direction of a translation (EPROP – ion engine)

- Solar cells are included in the mass of deployable and side panels

- Communication solution between the RU and the main spacecraft is not designed but assumed within 30 g (e.g., XBeeR chip)

- Telescope not yet designed but assumed that together with the framing camera is should have a mass of less than 1 kg

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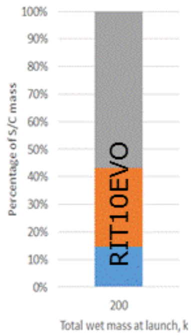
SS Inputs 2

- MA: Total Δv = 10.5 km/s
- EP:



- COMMS: M-Argo as reference for HGA
- GNC: JUICE as reference for NAVCAM

Mass Budget 2

Scaled from ESTCube-2 design

RIT10EVO double string – EPROP SS smaller tanks (L=601mm D=418mm V=60L)

19&22% of dry mass

		S/C [kg]		
	Mass Budget	**IEEE Pekka**	**Adpt.SPP2**	
RW 3 + Sun Sensor 6	Attitude, Orbit, Guidance, Navigation Control	0.29	7.80	SPP – 2.4kg
Patch + dipole + chip RU	Communications	0.19	3.00	M-Argo HGA
	Chemical Propulsion	0.00	0.00	
4 Bus PCB + 2 RU PCB	Data-Handling	0.48	3.60	
HV + tether + RU(reel + motor)	E-sail	0.45	0.00	
TILE 50 x5	Electric Propulsion	0.40	30.00	22% of dry mass
8cm telescope + 2cm FRA CAM	Instruments	1.00	7.00	JUICE NAVCAM
Hinges	Mechanisms	0.08	5.00	Smaller SA -> smaller SADM
SC 4BAT/4SD/5SP & RU 1BAT/2DP	Power	1.16	33.00	25% of dry mass
U-frame + bus/AOC + RU	Structures	0.90	20.00	15% of total dry mass
	System Engineering	0.00	0.00	
	Thermal Control	0.00	15.00	*Radiator: 0.4m2 -> 7.4kg*
	Harness 5%	0.25	6.22	
	Dry Mass w/o System Margin	**5.20**	**130.62**	
	Dry Mass w/ System Margin	**6.24**	**156.74**	
	EPROP Propellant Mass	0.00	57.97	27% of total mass (from RIT10EVO graph.)
	EPROP Propellant Residual	0.00	1.16	
	Total Wet Mass	**6.24**	**215.88**	

x35

A-SPP values from SPP OPT2 with corrections – no deployers

Power & Thermal Scaling Assumptions

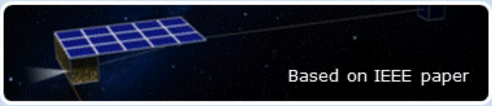
THERMAL

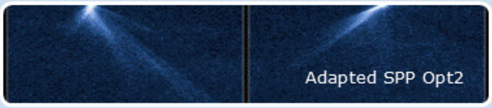
- Surface Area: 6 x 0.7m x 0.7m = 2.94m2
- MLI Losses: 250W / 11.7m2 x 2.94m2 = 60W (25% of SPP)
- Power Dissipation: (1500W x 6% + 40W) x (1 + 20%) = 160W
- Radiator Dissipation: 160W – 60W = 100W
- Radiator Area: From graph to dissipate 100W -> 0.4m2
- Radiator Mass: 37kg / 2m2 x 0.4m2 = 7.4kg (20% of SPP)
- Factor from Surface Area applied to all thermal equipment (25%)

POWER

- Solar Panels: 103.2kg x 1550W / 7150W = 23kg
- Battery and PCDU assumed 5kg each

System Design Summary



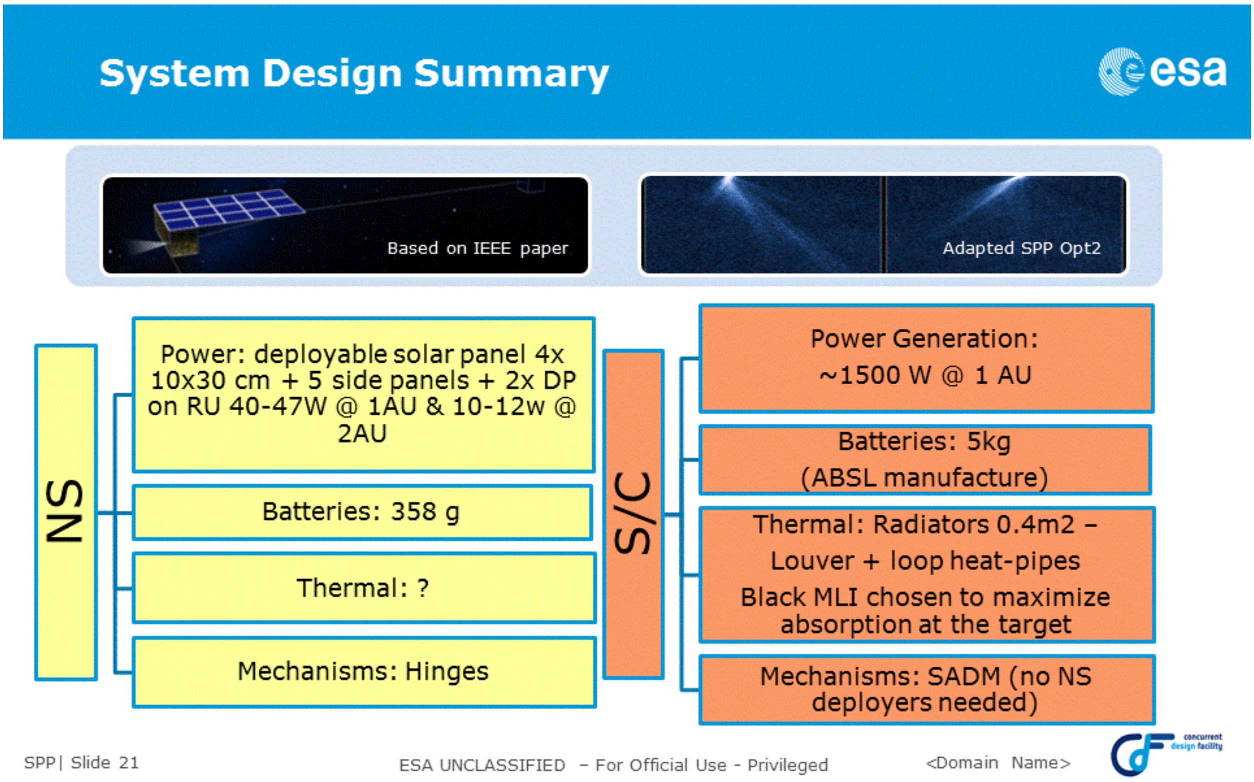


NS

- Total Mass: ~5kg
- Delta-V: 31.8km/s
- E-sail- 20 km tether + high voltage source + electron gun
  Input power: 7W @ 1AU (scales with 1/r2)
  Input voltage average: 15kV
- No propellant

S/C

- Total Mass: ~215kg
- Delta-V: 10.5 km/s
- Eprop: double string RIT10EVO
- Propellant mass: ~58kg


System Design Summary
esa
Based on IEEE paper
Adapted SPP Opt2
NS
Power: deployable solar panel 4x 10x30 cm + 5 side panels + 2x DP on RU 40-47W @ 1AU & 10-12w @ 2AU
Batteries: 358 g
Thermal: ?
Mechanisms: Hinges
S/C
Power Generation: ~1500 W @ 1 AU
Batteries: 5kg (ABSL manufacture)
Thermal: Radiators 0.4m2 – Louver + loop heat-pipes
Black MLI chosen to maximize absorption at the target
Mechanisms: SADM (no NS deployers needed)
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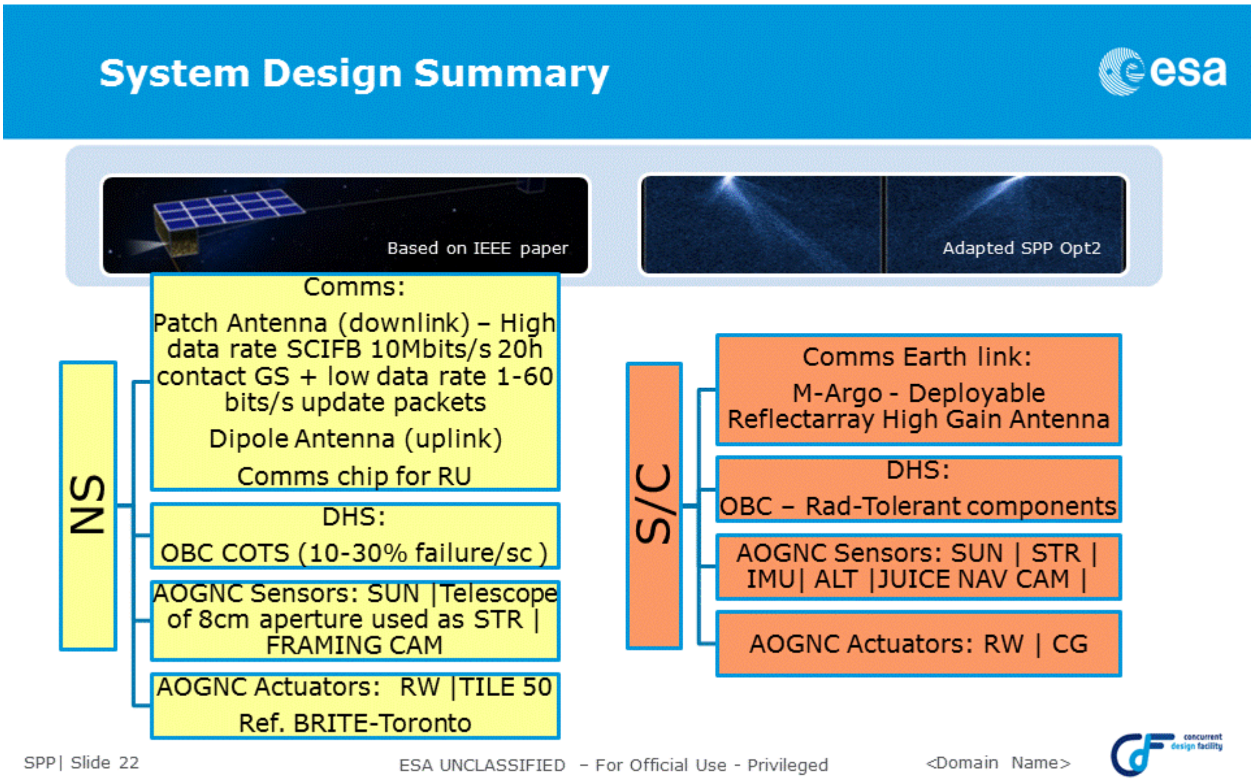
System Design Summary
esa
Based on IEEE paper
Adapted SPP Opt2
NS
Comms:
Patch Antenna (downlink) – High data rate SCIFB 10Mbits/s 20h contact GS + low data rate 1-60 bits/s update packets
Dipole Antenna (uplink)
Comms chip for RU
DHS:
OBC COTS (10-30% failure/sc)
AOGNC Sensors: SUN |Telescope of 8cm aperture used as STR | FRAMING CAM
AOGNC Actuators: RW |TILE 50
Ref. BRITE-Toronto
S/C
Comms Earth link:
M-Argo - Deployable Reflectarray High Gain Antenna
DHS:
OBC – Rad-Tolerant components
AOGNC Sensors: SUN | STR | IMU| ALT |JUICE NAV CAM |
AOGNC Actuators: RW | CG
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Mass Budget Discussion

Driving the "feasible" case → Communications

Mass Budget		S/C [kg] IEEE Pekka	Pekka Feas.	Adpt. SPP2	
Attitude, Orbit, Guidance, Navigation Control		0.29	0.29	7.80	
Communications		0.19	3.00	3.00	From M-Argo
Chemical Propulsion		0.00	0.00	0.00	
Data-Handling		0.48	0.48	3.60	
E-sail		0.45	1.79	0.00	4x 20 km Tethers
Electric Propulsion		0.40	1.58	30.00	
Instruments		1.00	7.00	7.00	JUICE NAVCAM
Mechanisms		0.08	0.08	5.00	
Power		1.16	5.00	33.00	
Structures		0.90	4.00	20.00	
System Engineering		0.00	0.00	0.00	
Thermal Control		0.00	5.00	15.00	
Harness	5%	0.25	1.41	6.22	
Dry Mass w/o System Margin		**5.20**	**29.63**	**130.62**	
Dry Mass w/ System Margin		**6.24**	**35.56**	**156.74**	
EPROP Propellant Mass		0.00	0.00	57.97	
EPROP Propellant Residual		0.00	0.00	1.16	
Total Wet Mass		**6.24**	**35.56**	**215.88**	

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Assessment

In red: finding/comments for ESA
In black: finding/comments for Pekka

Autonomy

- In the updated proposal the Nanosat is no longer fully autonomous, but transmission of telemetry is considered throughout the entire mission. The baseline mass budget of 5 kg shall be updated to reflect this functionality considered.

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Assessment – Mission Analysis

Review of Janhunen's paper and mission concept:

- Feasibility of conducting asteroid flyby sequence within the Main Belt with a thrust direction limited to 30deg from the radial direction, needs to be further assessed.
- Trajectories without thrust direction constraint to the asteroid belt can be optimized for significantly lower Delta-V that is more better suited for conventional EP missions
- Trajectories with primary and secondary targets add more constraints to the optimization problem (i.e. departure date) and should be look into more details
- Impact of Sun illumination phasing angle (especially while on second half of Main Belt tour) on the feasibility of detecting and tracking targets needs to be assessed.
- Mission Analysis is not considering size limitations for the encountered asteroids. This is a point for further assessment, closing the loop with the camera considered by GNC.
- In case of nanosats swarm each trajectory will be different, and the detectable targets would vary a lot. An extensive mission analysis would be required.
- Validating the autonomous on-board navigation function will require an enormous work.

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Assessment

EPROP

- Electron gun technology is feasible and could be derived from preexisting terrestrial technology, but needs to be developed to meet specific requirements (large performance range, space toughness, reliability, lifetime, ..) (TRP/GSTP activity w/ Finland/Estonia could help)
- Physical process to generate momentum is well understood; solar wind electric parameters well monitored; plenty of simulations of tether-based trajectories demonstrating its feasibility not only to accelerate towards solar wind direction, but also to brake. Still, no reliable data from any mission to verify validity of the models (STARS-II demonstrated faster de-orbiting with a single tether).
- TRL, however, in general is very low for the tether concept. Therefore, appropriate margins should be considered.
- Deployment concept considered to be the most challenging aspect. Substantial progress in the technology and suitable feasibility demonstration (e.g. in orbit) considered elemental to increase trust in concept (small-scale IOD helpful)

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Assessment

EPROP (ctd.)

- Mass budget based on many assumptions and idealisations. However, most of the spacecraft components are no different from a typical satellite. It should be more clearly identified, which components are off-the-shelf and/or with small modifications, and which components are completely new developments
- Propulsion system on remote unit to be defined more precisely with regards to power and fluidic management, thrust vector control, DOF. Method to stabilize tether after (and during) deployment with RU propulsion missing. However, propulsion system in remote unit only needed for multi-tether configurations. Other possibilities are to install steerable solar pressure sails. Single-tethered demonstration mission would reduce the need for propulsive subsystem in RU, as it can be stabilised with centrifugal force (i.e. cold gas in main satellite).

Assessment (Jesus)

GNC (1/2)

- It is assumed that nanosatellite is spinning, therefore some Time Delay Integration technique should be considered to obtain non-blurred images during detection phase and also during closest-approach (C/A).
  - Minimum exposure time from existing cameras shall be analysed for C/A
- For target imaging, the spin axis shall be rotated in order to have the telescope observing the target (power generation and thermal constraints to be considered).
- The need of RW in a spin-stabilized SC is not clear.
- Vision based navigation need is acknowledged, however the 150 km absolute and 10 km relative might not be needed and seem very challenging.
- Strategy to detect faint object (mag 13 and above) moving against the star background is far from trivial. Read-out, dark current, sky background and other noises affecting NAC VIS ASPECT are not addressed, their impact shall be carefully analyzed in order to achieve a reasonable SNR without too demanding pointing stability requirements.

Assessment (Jesus)

GNC (2/2)

- 5 days are assumed for the flybys. This number is strongly dependent on the apparent magnitude of the target (which depends on the absolute magnitude, distance to Sun, Sun phase angle, distance SC-asteroid) and the sensitivity of the assumed camera.
- Deeper analysis on the strategy to detect faint targets shall also consider the trajectory characteristics and the asteroid ephemerides error (typical values can be found on NASA horizons). It seems in general such long detection will not be available.
- The detection time shall be compatible with the delta-V required to compensate the B-plane deviation ('transversal' error), defined by asteroid ephemerides error and SC trajectory uncertainty (shorter detection times might be feasible depending of acceleration and relative error)
  - Note1: Rosetta camera was 25 kg and the fly-by velocity was 8.4 km/s
  - Note2: JUICE camera similar to ROSETTA is about 7.5 kg

Assessment (Jens)

- 6 days are assumed for the flybys, however the time needed for the observation is not taken into account. With 10 km/s approach velocity at 5 million km and the error that we are targeting (few m/s) 3 hours would be needed to detect the target against the star. Absolute size of the asteroids that can be observed might be limited, but also the velocity of the body and their albedo might represent a limitation of the observable bodies
- A typical relative velocity in the order of 10km/s and an expected pixel resolution of 18 m@1000km distance results in a maximum exposure time of ~2 ms (1 pixel smear acceptable). This is extremely challenging especially when considering the ultra low albedo (typically 0.06) of primitive asteroids. Active tracking is necessary.
- Note: Rosetta had a fly-by velocity of 8.6 km/s at Steins, followed by a successful imaging campaign. This could only be achieved by very complex S/C maneuvers (incl. autonomous asteroid tracking)planned long time ahead the encounter and a very advanced and resource hungry camera system (OSIRIS) and a resolution of only 80 m and the very high geometric albedo of the object of 0.39

Assessment

Thermal

- It is understood that the nanosatellite is spinning. This would make the thermal control rather challenging. It is unclear how the mass budget could have 0 kg allocation for the thermal design

On-board computer and mass memory

- **High mass memory reliability to ensure the long term archiving of the Science data until the Earth fly-by.**

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Assessment (Jens & Ana)

Instrument

- It is proposed to achieve a 17 m pixel resolution and spectral information in the NIR range by a multi spectral imager with 8 cm aperture and 2 m focal length, suggesting this will fit into 1U volume and 1 kg of mass.
- This goal seems rather very optimistic. Comparable camera systems (still with significantly lower focal length) flown world wide on planetary missions account for much larger resource budgets (ie 12-20 kg and 70U). This camera design will have a very small FOV, probably below 1 degree. This requires very precise pointing and knowledge of the target position. The proposed beam splitter required for the imager/spectrometer double function may reduce the S/N ratio significantly so that very long exposure times become necessary. This is not beneficial for analyzing the low albedo (primitive) asteroids
- **No instrument calibration based on Science data.**

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Assessment

Communication

- Autonomy at 3.15 is far from being an obvious link. The concept is in theory possible, however it shall be implemented on a well known platform with predictable behavior in order to reduce risks to acceptable level. The gap between physical principles and implementation is long and expensive

Assessment

- High number of SC (large swarm in deep space) is a low TRL concept. At the moment we cannot handle cheaply a swarm of satellites. Already ranging would heavily load the ground segment (Acquisition time for all of them is already substantial; and it depends on the link. At 3 AU this might require extra power on the platform and OPS concept needs to be carefully assessed)
- There is confidence that we could find ways to simplify however concepts and costs shall be carefully addressed.
- From the proposal it is understood that there is a RTU and the spacecraft, however it is not understood the communication strategy considered for the proposed system.

Assessment

General

- The overall deployment strategy of the nanosatellites shall be carefully assessed
- Solar Panel Mass per square meters considered in the proposal (102 g) is considered to conservative; thin film technology reach 1.5 kg per sqm. Moreover the spinning feature would imply a factor 2.3 to be accounted for.
- (We can compute the power that the nanosat cube could provide) to justify the lack of credibility of Pekka's proposal

Assessment

OPS

- The OPS concept shall be elaborated, in particular with respect to the needs of the novel eSail propulsion strategy.
- The **GS&Ops** concept defined for SPP would not be applicable (higher FD**,** MA and **Control Team** support **will be needed, feasibility to fulfill the ground station coverage of the feet to be analysed, mission control system should be adapted for multi-mission support**)

It is thought that the EO mega-constellation Concept of Operations is of no use in this scenario: high number of ground stations, EO nanosats allow high level of ground automation, etc. These are some examples on the ground segment, it should be added the adaption of the space segment.

Comments on Communications Design 1/2

- Spacecraft design, including communications system, should require deeper analysis. " *During the mission, the communication system is planned to be used in two different modes: 1.Telemetry downlink: Using low data-rate communications with 1 bit/s to 60 bit/s status update packets will be sent to the Earth. This will be either scheduled by the on-board computer or requested from the Earth"*

1-60bps is not acceptable, low bit rates are for emergency cases only (Safe: ~60bps TBD), see COMM presentations.

Telecommand rates must cope with operational needs during the mission, or the other way around.

Comments on Communications Design 2/2

- *" 2. High data-rate science data downlink: The system should be able to achieve 10 Mbit/s downlink speed during a flyby, which would allow to transmit science data in 20-hour communication window with DSN receivers or other DSNcompatible antennas."*

    vs. 2016 proposal of

    "the deepspace network time needed per spacecraft is of order 3 hours only"

-> Numbers to be review and impact to be analized.

- *" For uplink, a dipole antenna can be utilized. As DSN offers high power uplink capabilities, then simple dipole antenna can receive enough power from the Earth inside the main belt. "*

-> Clarify *"high power uplink capabilities"*

Conclusion

- We believe that the concept is feasible with a classical EP solution, however the spacecraft would not be a nanosatellite
- The tether concept works in theory, however the implications of designing a full platform within the boundaries of a small mission are not obvious
- If we reuse MARGO comms, we would have the limitation coming from the distance from the Sun (max 2 AU with standard comms 500 bps at best – it might go down to 100 bps). MARGO would suit the semi-autonomous concept. AIM had 1-2 kpbs at 3 A.U. (mimimal but standard. Not sufficient for science data download at 3 A.U. but possible on the way back to Earth)
- The autonomy of a small platform (still not 5 kg) comes along with a huge risk. A trade-off between a high number of satellites versus one bigger reliable mission shall be performed. (In Bepi Colombo there are 2 OBH in order to secure the ephemeris in case of 1 OBC failure)

Disclaimer

- The SPP Nanosat concept has been revisited in order to implement the MultiAsteroid mission concept. It has to be noted that the Nanosat Mission considered in the SPP Study is based on requirements different from those considered in the Proposal from P.J. (different PL/ science data relay strategy are completely different, just to mention a few)